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DRL No. T-747
Line Item 3
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FINAL REPORT

EARTH ORBIT NAVIGATION STUDY

VOLUME 2

SYSTEM EVALUATION

August 25, 1972

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

TRW
SYSTEMS GROUP

HOUSTON OPERATIONS • P. O. BOX 58327, HOUSTON TEXAS 77058 • (713) 333-3133

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Approved by: R. H. Kidd
R. H. Kidd
Project Manager

TRW
SYSTEMS GROUP

ACKNOWLEDGEMENT

The contents of this document were developed from the contributions of the following people:

J. E. Earl

W. W. Ferry

J. L. Knoedler

W. G. Marley

R. J. Panneton

L. H. Robinson

I. J. Williams

ABSTRACT

The Earth Orbit Navigation Study, Contract No. NAS 9-12475, was conducted by TRW Systems Group of TRW Inc., for the Manned Spacecraft Center (MSC) of the National Aeronautical and Space Administration (NASA). Period of performance on the contract was 25 February 1972 to 25 August 1972.

The objective of the study was to perform an overall systems evaluation of five candidate navigation systems in support of earth orbit missions. These five systems are horizon sensor system, unknown landmark tracking system, ground transponder system, manned spaceflight network, and tracking and data relay satellite system. Two reference missions were chosen: a low earth orbit mission and a transfer trajectory mission from low earth orbit to geosynchronous orbit. The specific areas addressed in the evaluation were performance, multifunction utilization, system mechanization, and cost.

This final report consists of two volumes. Volume I, an Executive Summary Volume, contains an overview of the study. Volume II contains the detailed results of the evaluation of the five navigation systems.

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ILLUSTRATIONS (continued)

ILLUSTRATIONS (continued)

NOMENCLATURE

ALS	Shuttle vehicle
AROD	Advanced range and orbit determination system
BINOR	Binary optimum ranging
BW	Bandwidth
CEA	Correlation electronics assembly
CMG	Control moment gyro
CW	Continuous wave
DB	Decibel
DDT&E	Design, development, testing, and engineering
DELTRA	Dedicated landmark tracker
DOD	Department of Defense
DSN	Deep space network
EPR	Effective radiated power
ETR	Eastern test range
EVA	Extra vehicular activity
FFM	Free flying modules
FM	Frequency modulation
FOV	Field of view
GHZ	Giga hertz
GRARR	Goddard Range and Range Rate
GSFC	Goddard Spaceflight Center
HZ	Hertz
IR	Infrared
KHZ	Kilo hertz
KWH	Kilowatt hours
Lb	pound

LOS	Line of sight
MDC	McDonnell Douglas
MHZ	Mega hertz
MMOS	Multimode Optical sensor
MSC	Manned Spacecraft Center
MSFN	Manned Spaceflight Network
MTBF	Meantime between failure
NA	Not applicable
NAR	North American Rockwell
NASA	National Aeronautical and Space Administration
NASCON	NASA Communications Network
NM	Nautical miles
PADS	Precision attitude determination system
PPM	Parts per million
PN	Pseudo noise
PRN	Pseudo random noise
PWR	Power
R	Range
$\dot{R}$, RR	Range rate
RAM	Research Application module
RCVR	Receiver
RF	Radio frequency
RMS	Remote maneuvering subsatellite
RSS	Root sum square
SEA	Sensor electronics assembly

NOMENCLATURE (Continued)

SGLS	Space guidance and landing system
SGU	Sensor gimbal assembly
SITU	Special interface and timing unit
SLANT	Star/landmark tracker
S/N	Signal to noise ratio
SS	Space station
SSA	Star sensor assembly
STADAN	Satellite Tracking and Data Acquisition Network
TBD	To be determined
TDM	Time division multiplexing
TDRS	Tracking and Data Relay Satellite
USB	Unified S-band
UV	Ultraviolet
VHF	Very high frequency
XMITTER	Transmitter
XMT	Transmit

INTRODUCTION

The Earth Orbit Navigation study, Contract No. NAS 9-12475, was conducted by TRW Systems Group of TRW Inc. for the Manned Spacecraft Center (MSC) of the National Aeronautical and Space Administration (NASA). Period of performance on the contract was 25 February 1972 to 25 August 1972.

The objective of this study was to perform an overall systems evaluation of candidate navigation systems in support of earth orbital missions. Two missions were considered - a low earth orbit mission and a transfer mission from low earth orbit to geosynchronous orbit. The space station characterization of the low earth orbit mission was chosen because of the data available, the long term duration, the potential crew involvement, and its potential as a future NASA program option. The tug vehicle transfer orbit presents problems unique to the vehicle and the large range in operating altitudes. The five candidate navigation systems considered were:

Manned Spaceflight Network (MSFN)

Tracking and Data Relay Satellite (TDRS)

Ground Transponder System

Unknown Landmark Tracking System

Horizon Sensor System

The capabilities of these systems to support the functional requirements of the two reference missions was evaluated by addressing the four tasks defined in Section 3 of the contract statement of work.

TASK 3.2.1 ORBIT NAVIGATION ACCURACY

Investigate the performance capabilities of the navigation systems considering real world effects significant to navigation accuracy and identify the characteristics limiting system performance.

TASK 3.2.2 MULTIFUNCTION UTILIZATION

Investigate modifications of the basic navigation system to utilize the full potential of the system to provide information for other mission and vehicle functions.

TASK 3.2.3 MECHANIZATION

Perform top level mechanization studies to define a basic mechanization scheme and to identify problem areas or special requirements.

TASK 3.2.4 COST SENSITIVITY

Evaluate the relative costs of the navigation systems considering those parameters which have a major influence.

The final report of this study is comprised of two volumes. This volume provides the detailed discussion of the four tasks described above.

SUMMARY

NAVIGATION PERFORMANCE

The navigation performance and characteristics of the candidate systems for low earth orbit application are presented in Table 2-1, showing steady state accuracies, error convergence characteristics, and limiting factors for each system. The data presented represents navigation on a circular orbit of approximately 250 nautical mile altitude and 55 degree inclination.

Each system achieves convergence to steady state navigation performance within one revolution of tracking (provided the coverage is available) except for the horizon sensor system which requires up to two revolutions of data. The systems generating range and range rate data (ground transponders, MSFN, and TDRS) have similar navigation characteristics, each providing about 1000 feet (1σ) accuracy. The least accurate of the systems considered is the horizon sensor system, capable of approximately a 4500 foot (1σ) navigation accuracy and limited mainly by the horizon altitude uncertainty. The navigation performance of the landmark tracker system (2000 feet 1σ) is dependent on the availability of sunlit land masses each revolution.

The unknown landmark tracking system and the ground transponder system cannot support the tug transfer mission from low earth orbit to geosynchronous orbit because of their altitude and range limitations. The navigation performance of the horizon sensor system, the TDRS, and MSFN for the transfer trajectory are summarized in Table 2-2. The navigation uncertainty is presented at the nominal time of arrival at geosynchronous altitude (termed intercept).

The MSFN system navigation accuracy converges with one hour of tracking from a single station to 16,000 feet at intercept, and then reduces to 2200 feet when data from a second station is incorporated one half way through the trajectory. With tracking from a single station, the dominant error is the out-of-plane component.

The TDRS system provides tracking data only during the first hour of the transfer trajectory because of the coverage of the relay satellite antenna (30 degrees). The performance of the system (26,789 feet at intercept) represents tracking from a single TDRS; additional tracking from a second TDRS did not improve the performance. The inferior performance of the TDRS as compared with MSFN is apparently the result of a smaller change in the measurement geometry.

The horizon sensor system performance is based upon tracking at selected 15 minute intervals during the entire transfer trajectory. The navigation error at intercept slowly improves as additional data is incorporated until intercept occurs with a navigation uncertainty of 41,000 feet. The major error contribution for this system is the horizon altitude uncertainty.

MULTIFUNCTIONAL USAGE

The functional requirements defined for the space station which can be considered for multifunctional usage of the five navigation systems are:

Table 2-1. Low Earth Orbit Navigation Performance Comparison

SYSTEM	ORBIT NAVIGATION ACCURACY (1σ)	TRACKING REQUIREMENTS FOR ERROR CONVERGENCE	LIMITING FACTORS
Ground Transponder	1000 ft	Tracking data from two beacons with a minimum separation of 1/3 revolution.	Inplane error growth during periods of non-coverage.
Horizon Sensor	4500 ft	1.5 to 2 revolutions of sensor data.	Unmodeled variations in the horizon altitude.
TDRS	1000 ft	Tracking data from two TDRS over 1/2 an orbit revolution.	TDRS ephemeris errors.
MSFN	900 ft	Tracking data from two stations with a minimum separation of 1/2 revolution.	Inplane error growth during periods of non-coverage.
Landmark Tracker	2000 ft	Tracking data from five landmarks over 1 revolution.	Non-coverage due to cloud cover. Sensor pointing errors.

Table 2-2. Transfer Orbit Navigation Performance Comparison

	NAVIGATION ERROR AT NOMINAL TIME OF ARRIVAL AT GEOSYNCHRONOUS ALTITUDE							
	POSITION ERROR - 1σ (FEET)				VELOCITY ERROR - 1σ (FPS)			
NAVIGATION SYSTEM	U	V	W	RSS	U	V	W	RSS
Propagation without updates	112,531	162,851	12,172	198,323	16.9	3.8	.3	17.3
Horizon Sensor System	26,443	29,298	11,133	41,007	3.4	.9	.4	3.6
TDRS	20,204	12,077	12,791	26,789	2.4	.6	.2	2.5
MSFN	83	2,035	788	2,184	.07	.02	.02	.08

Coordinate System:

- U - Radial
- V - Downrange
- W - Crossrange

- Communications
- Rendezvous and stationkeeping navigation (range from 1000 feet to 1100 nautical miles)
- Attitude reference
- Functional redundancy

The potential capabilities of the systems to support these functions are as follows:

MSFN and TDRS - As an integral part of the communication system, these systems will perform this function. Systems do not provide any additional functional support.

Ground Transponder - System can support rendezvous with range and range rate measurements but must use a gimballed antenna or an optical device to provide angle measurements required for rendezvous. Some functional redundancy is available during rendezvous from the comparison of the range, range rate, and angle measurements.

Horizon Sensors - Can provide attitude reference redundancy in two axes in a local vertical coordinate system; with a state vector and one gyro, the complete inertial attitude can be established. Internal redundancy is provided by a four sensor head configuration.

Landmark Tracker - Proposed mechanization would use same optics for attitude reference and navigation. System can support rendezvous with angle measurements on a flashing beacon, but needs a range measurement in addition to complete the required data set.

MECHANIZATION

The TDRS and MSFN transponders which must be integrated with the communication system will be one of three types depending upon the ranging technique used by the system:

Tone - Fixed frequency tones such as used on Apollo VHF ranging and Goddard range and range rate tracking and telemetry system.

Digital Code - Digital ranging techniques such as used by Apollo unified S-band, and JPL Deep Space Network.

Super Sync - Combined ranging and telemetry signal technique which provides much faster acquisition with less total power.

The most likely candidate is the pseudo random noise (PRN) technique used on Apollo with the transponder elements including a phase locked receiver, a coherent translator, a PRN code generator, and an oscillator.

Two proposed ground transponder systems can support low earth orbit missions - the CUBIC CR100-4 and the MOTOROLA AROD systems. The CR100-4 system is favored over the AROD system because it is in a more advanced state of development and testing and it is cheaper. The additional capability of AROD system to interrogate four stations simultaneously offers no real advantage for orbital application when a minimum number of ground stations are employed. The maximum range of 1500 nautical miles for these systems apparently cannot be extended without a major redesign of the system. Thus, the system cannot be considered as a viable candidate for the tug geo-synchronous mission. The low system reliability (relative to the other candidate systems) represents a problem in maintenance and replacement for the space station mission.

The tradeoffs available for a horizon sensor system indicate a definite all around advantage of a mechanization approach utilizing a strapdown star tracker with the Quantic Mod IV horizon sensor system. This system is definitely preferable for the space station which maintains either local vertical or inertial hold attitude modes. A multimode optical sensor offers desirable advantages for a maneuverable spacecraft such as the tug. However, the limitation of taking measurements only on the sunlit horizon places a severe restriction on mission planning for high altitude orbits. The potential problems of this system are 1) the relative alignment between the attitude reference package and the horizon sensor package and 2) variation of sensor horizon altitude measurement as a function of spacecraft altitude and attitude on the tug transfer trajectory.

A design for an automatic unknown landmark tracker has been developed based on the use of an image dissector which provides an electronic means for scanning the earth scene. Two tracker designs have been developed. The Dedicated Landmark Tracker (DELTRA) uses the NASA/TRW PADS reference gimbal design to provide adequate gimbal freedom for unknown landmark tracking. A Star/Landmark Tracker (SLANT) design uses an enlarged gimbal freedom to allow for the tracking of both stars and unknown landmarks with the same tracker on a time-shared basis. The required adjustment in sensitivity is accomplished by changing the high voltage to the image dissector between alternate tracking periods. Alternate mechanizations of the star sensing and landmark tracking sensors were considered to determine the optimum approach for the system. The strapdown SPARS star sensors and the gimballed PADS star tracker were considered along with the DELTRA landmark tracker and integrated tracker approaches including the SLANT. The tradeoff study shows a definite advantage for the SLANT mechanization particularly with regard to weight, power, size and reliability.

COST ANALYSIS

A comparison of the potential cost of implementing each of the five candidate systems for the earth orbit navigation function was developed to account for the different areas in which expenditures will be necessary and to account for the various factors which can affect the cost.

The implementation costs throughout the duration of each program were divided into two phases:

- Design, Development, Testing, and Engineering (DDT&E) - cost for system development
- Investment and Operations (I&O) - cost associated with purchase of production systems and mission operations

These two cost categories are distinct in that DDT&E costs are once only costs, and I&O costs are recurring costs depending heavily upon the number of missions and vehicles using the navigation system. Operational costs for MSFN and TDRS systems do not include the costs of maintaining these systems.

The operational mission and program characteristics also affect the system cost through the number of scheduled missions and the duration of each mission. The two vehicles and programs considered were the space tug with 126 flights over a six year period and the space station with a mission length of ten years. The total functional requirements of each vehicle were considered in terms of the multifunctional capability of each candidate system.

Tables 2-3 and 2-4 summarize the relative program costs of implementing each of the five candidate systems for the space tug and space station programs respectively. The costs were computed relative to an arbitrary base system which was taken as the star/landmark system consisting of a combined star/ landmark tracker (which also provides line-of-sight angle data for rendezvous and stationkeeping navigation) and a communications receiver/transmitter (which also provides ranging for rendezvous and stationkeeping navigation). Since a definite tradeoff exists for the rendezvous and stationkeeping navigation sensor (communication ranging addition combined with an optical tracker versus a separate rendezvous radar), the cost of implementing each candidate system with each rendezvous and stationkeeping navigation sensor was determined. These data show that rhe rendezvous radar option is more expensive than the optical tracker option for all candidate navigation systems except the ground transponder system.

Inspection of Tables 2-3 and 2-4 shows that the ground transponder system is the most expensive system to implement (relative to the other four candidate systems) for either rendezvous and stationkeeping navigation sensor on either the space tug or space station program. This is due primarily to the poor reliability of both the onboard interrogator and the ground transponder resulting in a large purchase of production units and expensive ground checkout and system repair.

When the rendezvous radar is chosen as the rendezvous and stationkeeping navigation sensor, the least expensive system to implement on either the tug or space station would be the star/landmark tracker system. The tracker for this system can provide star and rendezvous target line-of-sight angle data and therefore only a communication ranging addition is necessary to provide all the required functions. As a result the other four systems are severely penalized by the cost of adding a separate rendezvous radar where the star/landmark tracker system is not. This separate rendezvous radar penalty is not only significant in the relative development cost but also very predominant on the tug in the cost of weight.

Table 2-3. Summary of Costs Relative to the Star/Landmark Tracker System for the Tug Program

SYSTEM / COST ELEMENT	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK TRACKER SYSTEM
DDT&E (13)	3.0 (-0.3)	2.5 (-0.8)	2.0 (-0.6)	2.5 (-0.8)	0 (0)
GROUND STATION INSTALLATION	NA	NA	0.1 (0.1)	NA	NA
SOFTWARE	0 (0)	-0.2 (-0.2)	0 (0)	-0.2 (-0.2)	0 (0)
SUBTOTAL	3.0 (-0.3)	2.3 (-1.0)	2.1 (-0.5)	2.3 (-1.0)	0 (0)
PRODUCTION UNITS (13)	0.7 (0)	0 (-0.6)	0.9 (1.4)	0 (-0.6)	0 (0)
WEIGHT (6)	5.7 (1.9)	3.7 (-0.1)	3.1 (1.8)	3.7 (-0.1)	0 (0)
POWER (8)	0.1 (0.1)	0 (0)	0.6 (0.7)	0 (0)	0 (0)
GROUND CHECKOUT AND SYSTEM REPAIR (9) (10)	0.6 (0.1)	0.4 (-0.1)	1.3 (1.1)	0.4 (-0.1)	0 (0)
RELIABILITY (12)	0.7 (0.6)	0.7 (0.6)	4.9 (9.2)	0.7 (0.6)	0 (0)
SUBTOTAL	7.8 (2.7)	5.0 (-0.2)	10.8 (14.2)	5.0 (-0.2)	0 (0)
MISSION GROUND SUPPORT	NA	2.8 (2.8)	NA	2.8 (2.8)	NA
GROUND STATION OPERATION AND MAINTENANCE	NA	-	0.6 (0.6)	-	NA
TOTAL	10.8 (2.4)	10.1 (1.6)	13.5 (14.3)	10.1 (1.6)	0 (0)

NOTES:
1. Parenthetical numbers are for a ranging (Com Δ) and an optical tracker for rendezvous; open numbers are for a radar for rendezvous.
2. All numbers are in millions of dollars.
3. Numbers under Cost Element give corresponding table number in Cost Analysis Section.

Table 2-4. Summary of Costs Relative to the Star/Landmark Tracker System for the Space Station Program

SYSTEM / COST ELEMENT	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR LANDMARK TRACKER SYSTEM
DDT&E (13)	3.0 (-0.3)	2.5 (-0.8)	2.0 (-0.6)	2.5 (-0.8)	0 (0)
GROUND STATION INSTALLATION	NA	NA	0.1 (0.1)	NA	NA
SOFTWARE	0 (0)	-0.1 (-0.1)	0 (0)	-0.1 (-0.1)	0 (0)
SUBTOTAL	3.0 (-0.3)	2.4 (-0.9)	2.1 (-0.5)	2.4 (-0.9)	0 (0)
PRODUCTION UNITS (13)	0.6 (0)	0.3 (-0.2)	2.8 (2.5)	0.3 (-0.2)	0 (0)
WEIGHT (7)	0.1 (0)	0.1 (0)	0.3 (0.1)	0.1 (0)	0 (0)
POWER (8)	0.1 (0.2)	-0.1 (0)	0.8 (0.9)	-0.1 (0)	0 (0)
GROUND CHECKOUT AND SYSTEM REPAIR (10)	0.3 (-0.1)	0.3 (-0.1)	2.3 (1.8)	0.3 (-0.1)	0 (0)
RELIABILITY (11)	0.1 (0)	0.1 (0)	0.2 (0.3)	0.1 (0)	0 (0)
SUBTOTAL	1.2 (0.1)	0.7 (-0.3)	6.4 (5.6)	0.7 (-0.3)	0 (0)
MISSION GROUND SUPPORT	NA	6.3 (6.3)	NA	6.3 (6.3)	NA
GROUND STATION OPERATION AND MAINTENANCE	NA	-	1.0 (1.0)	-	NA
TOTAL	4.2 (-0.2)	9.4 (5.1)	9.5 (6.1)	9.4 (5.1)	0 (0)

NOTES:
1. Parenthetical numbers are for a ranging (Com Δ) and an optical tracker for rendezvous; open numbers are for a radar for rendezvous.
2. All numbers are in millions of dollars.
3. Numbers under Cost Element give corresponding table number in Cost Analysis section.

For the case where a communication ranging addition combined with an optical tracker is chosen as the rendezvous and stationkeeping navigation sensor, the TDRS, MSFN and star/landmark systems are the least expensive and would cost approximately the same to implement on the space tug. The horizon sensor system would cost an additional 2-3 million dollars to implement on the tug. For the space station program the mission ground support costs for TDRS and MSFN become quite large due to the length of the program and the amount of navigation support required. As a result, the star/landmark tracker and the horizon sensor systems would be the cheapest to implement on the space station.

CONCLUSIONS AND RECOMMENDATIONS

The results obtained from the evaluation of the five navigation system are summarized in the conclusions and recommendations listed below.

1) The ground transponder system represents a fine navigation system in terms of accuracy, coverage, and growth potential. However, this system is very expensive and apparently cannot support a geosynchronous orbit mission without a major redesign of the system. The limiting factor in application of this system for a long term mission is its low reliability.

2) The unknown landmark tracking system is the cheapest system considered for both the tug and space station programs because of its multifunctional capabilities. However, the system cannot support a geosynchronous orbit mission because of its altitude limitations. The system is a very good candidate for space station application.

3) The horizon sensor system is competitive costwise and can support both the low earth orbit and the high earth orbit missions. Although, the navigation accuracy attainable with this system is somewhat marginal and has a limited growth potential, the reliability, simplicity, and builtin redundancy of this system make it a very attractive candidate for the space station.

4) The TDRS system is an expensive system for space station application but is cost competitive for the tug program. The capability of the system to support a tug mission is limited because of the 30 degree coverage constraint of the satellite antenna. However, if the ground tracking stations dedicated to the satellites can be used to track the tug, then the TDRS system will be the best system considering communications, accuracy and operational cost.

5) The MSFN system is expensive for space station application, but is the only candidate which can be considered completely adequate for the tug mission. This system will fulfill the communication requirements in addition to the navigation requirements, but has a very high operational cost.

6) The capability of the horizon sensor system to support a tug high orbit mission should be evaluated with particular emphasis on horizon modeling, sensor modeling, and software development. This analysis should be performed with an engineering simulation.

7) The capability and cost effectiveness of the unknown landmark tracking system for low earth orbit application warrants further support for hardware development and testing.

PERFORMANCE ANALYSIS

The orbit navigation systems were evaluated to determine their navigation capability on two orbital missions: low earth orbit and a transfer orbit from low earth orbit to geo-synchronous orbit. The discussion of each system includes a general description of the system operation for each mission, a comprehensive error model for the navigation measurements and a summary of the orbital navigation accuracy. The results presented here are intended to provide a single data source for comparative evaluations of the navigation performance.

LOW EARTH ORBIT PERFORMANCE

The low earth orbit trajectories considered orbital altitudes between 200 and 300 nautical miles and inclinations between 35 and 90 degrees. The typical trajectory used was 270 nautical miles altitude and 55 degrees inclination.

The following environmental errors were considered in this evaluation.

- Drag effects:

  $C_D A/2m = 0.118\ ft^2/slug$

  Error (1σ) = 10% (or .012 $ft^2/slug$)

- Gravitational potential:

  $\mu(1\sigma)$ error 2 ppm or equivalent error which yields about 500 feet per revolution.

- Speed of Light Uncertainty (1σ) = 0.5 ppm

- Initial state error covariance matrix in feet, feet per second units and in a radial, downrange, and crossrange coordinate system (typical shuttle injection propagated one half revolution):

$$\begin{bmatrix} 2.353+7 & -1.005+8 & -1.961+7 & 9.893+4 & -2.377+4 & 3.558+4 \\ & 5.940+8 & 3.901+7 & -5.481+5 & 1.112+5 & -7.046+4 \\ & & 1.053+8 & -4.792+4 & 1.736+4 & -1.912+5 \\ & & & 5.117+2 & -1.073+2 & 8.671+1 \\ \text{SYMMETRIC} & & & & 2.459+1 & -3.147+1 \\ & & & & & 3.472+2 \end{bmatrix}$$

Ground Transponder Navigation System

The ground transponder navigation system considered here consists of an interrogator (transmitter/receiver) located onboard the orbiting vehicle and a set of ground based transponders placed at convenient locations on the earth. The interrogator transmits a signal that is received by the transponders and retransmitted. The return signal received by the interrogator is processed onboard the orbiting vehicle to obtain a measurement of range and range rate relative to the ground transponder. The range data is obtained by measuring the phase shift in the signal resulting from propagation to and from the transponder. The range rate data is obtained by measuring the Doppler shift in the received signal frequency.

Selection of the ground station locations must be based upon the orbits under consideration and the desired navigation accuracy. The navigation accuracy achievable when tracking a ground transponder will meet the most demanding navigation requirements. The primary factor in determinimg an acceptable set of station locations is the longest period of non-coverage that can be allowed before the navigation error growth exceeds the requirements. Other considerations that influence the choice of station locations are:

(1) The number of stations required should be minimized to reduce the cost of installation and maintenance;

(2) The location of all stations within U. S. territory is highly desirable;

(3) The location of stations at operating airfields or military bases is highly desirable to facilitate maintenance.

A candidate set of transponder locations obtained from Reference 19 is presented in Figure 3-1. The seven stations are located in U. S. territory and provide adequate coverage except for extremely low inclination orbits (Reference 19). For a 270 nautical mile circular orbit at 55 degree inclination, the maximum interval of non-coverage is approximately 2.5 revolutions. For a 200 nautical mile orbit at 90 degree inclination, the maximum interval of non-coverage is approximately 1.75 revolutions. This set of stations also provides adequate coverage for all proposed space shuttle mission orbits.

System Error Model

The error in the range and range rate measurements from a ground transponder system can be attributed to the following error sources:

(1) interrogator and transponder equipment error;

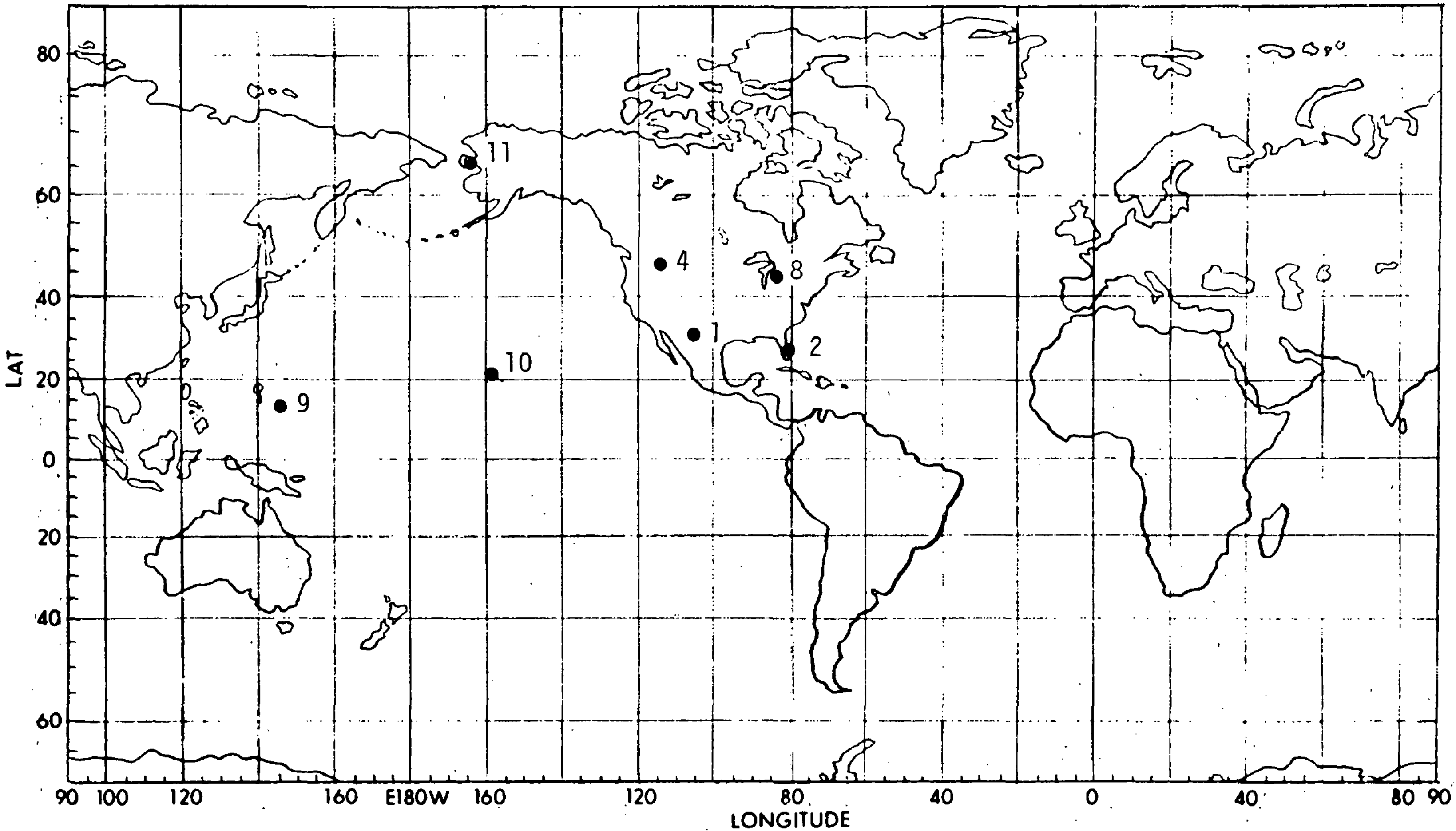
(2) multipath error in the ground-to-air link;

(3) error in the refraction correction;

(4) station location error.

An error budget for the equipment error in a ground transponder system operating at orbital altitudes is presented in Table 3-1.

The phenomenon known as multipath reception arises when the signal is received from more than one propagation path by reflection from the ground or nearby objects. The random error caused by multipath effects is a



1. Biggs AFB, Texas (31.85°N. Lat., 106.38° W. Long.)
2. Cape Kennedy (28.65° N., 80.57°W.)
4. Fairchild AFB, Washington (47.8° N., 117.5° W.)
8. Wurtsmith AFB, Michigan (44.5° N. Lat., 83°W. Long.)
9. Anderson AFB, Guam (13.58° N., 144.92°E.)
10. Honolulu Intl. (21.33°N., 157.92°W.)
11. Alaska (65°N., 165°W.)

Figure 3-1. Ground Transponder Locations (Reproduced from Reference 19)

Table 3-1. Ground Transponder System Error Model (1σ)

RANGE MEASUREMENT		
Random Error		
Signal to noise		1.0 ft
Phase shift over dynamic range		1.0 ft
Phase shift over temperature range		1.0 ft
System error due to craft dynamics (25,000 fps, 1,000 ft/sec^2)		0.2 ft
Multipath		3.0 ft
Digitization		0.3 ft
	RSS	3.5
Bias Error		
Calibration		1.0 ft
Oscillator stability	Scale Factor	0.1 PPM
Velocity of light	Scale Factor	0.5 PPM
RANGE RATE MEASUREMENT (.9 sec. count interval)		
Random		
Signal to noise		.01 ft/sec
System error due to craft dynamics		.001 ft/sec
Digitization error		.014 ft/sec
Multipath		.01 ft/sec
	RSS	.02 ft/sec *
Bias		
Oscillator stability	Scale Factor	1.0 PPM
Velocity of light	Scale Factor	0.5 PPM

*Cubic has obtained measured RMS value of .06 ft./sec.

function of the elevation angle (maximum at 90 degrees) and the system mechanization (Reference 30).

Reference 9 gives the following values for multipath effects:

$$\left.\begin{array}{l}\text{Range} = 3 \text{ ft} \\ \text{Range Rate} = .01 \text{ fps}\end{array}\right\} 1\sigma$$

Atmospheric refraction causes the signal to propagate along a curved path between the interrogator and transponder. The shape of the curved path and the velocity of propagation are determined by the atmospheric index of refraction. An onboard model of the refraction effects can be used to make corrections to the range and range rate measurements to compensate for deviations caused by atmospheric refraction. Since the refractive index varies with pressure, temperature, and water vapor content, the accuracy that one can determine these variables limits the ultimate accuracy of the refraction corrections. The deviation in range and range rate measurements due to refraction is presented in Figures 3-2 and 3-3 as a function of elevation angle for a circular 250 nautical mile orbit. The curves assume a horizontally homogeneous exponential atmosphere model with a surface refractivity N_S value indicated on the curves. The curves clearly show the significant increase in the refraction effect for elevation angles below 10 degrees. Also, for angles above 10 degrees, the refraction correction is insensitive to variations in surface refractivity. The residual error in the range and range rate data after refraction corrections have been applied is due to variations in the surface refractivity N_S and deviations of the true atmosphere from the exponential model. The residual errors increase for decreasing elevation angles. An estimate of the 1σ residual measurement errors for a 5 degree elevation angle is (Reference 14):

Range - 18 ft

Range Rate - .3 fps

Conservative values for station location errors are obtained by assuming values slightly larger than the MSFN location uncertainties. These 1σ location uncertainties are:

Δ altitude = 200 ft

Δ latitude = 2 $\widehat{\text{sec}}$

Δ longitude = 2 $\widehat{\text{sec}}$

Navigation Performance

The steady state navigation performance for the ground transponder system is characterized by small position errors during intervals of transponder tracking followed by growth of the inplane error during periods of non-coverage. Representative steady state inplane and crosstrack position errors for navigation on a 256 nautical mile circular orbit are presented in Figure 3-4.

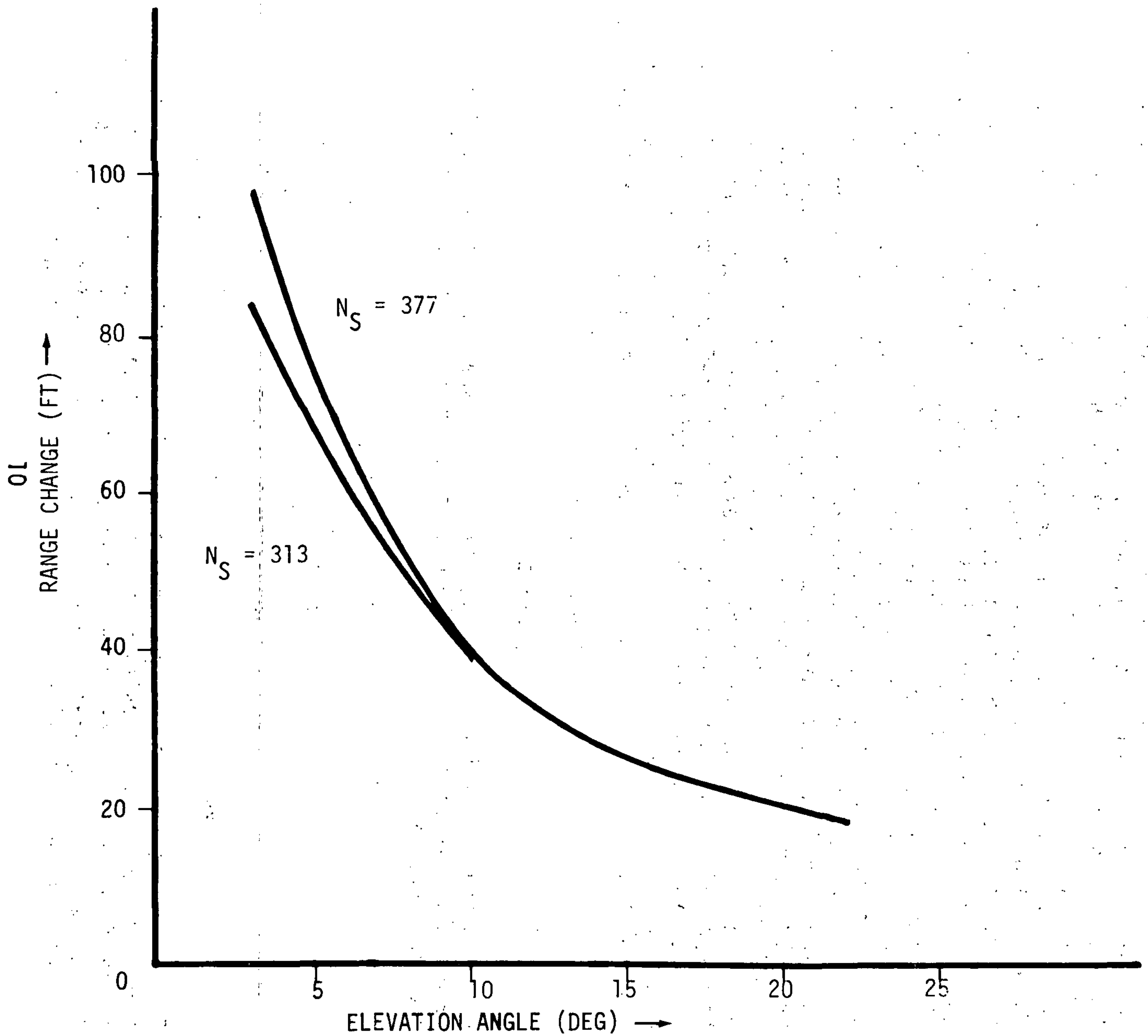


Figure 3-2. Change in Range Due to Tropospheric Refraction for 250 N.Mi. Orbit

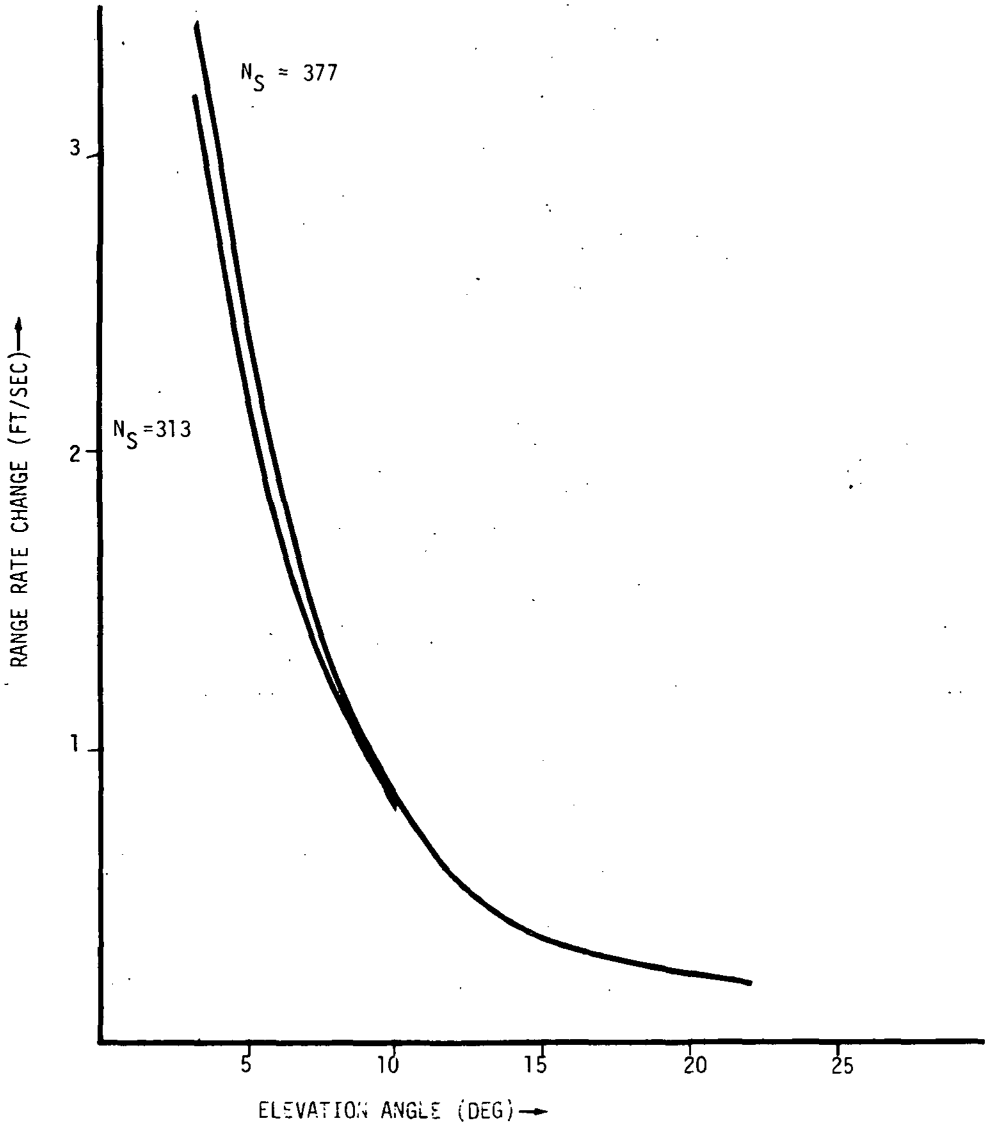


Figure 3-3. Change in Range Rate Due to Tropospheric Refraction for 250 N.Mi. Orbit

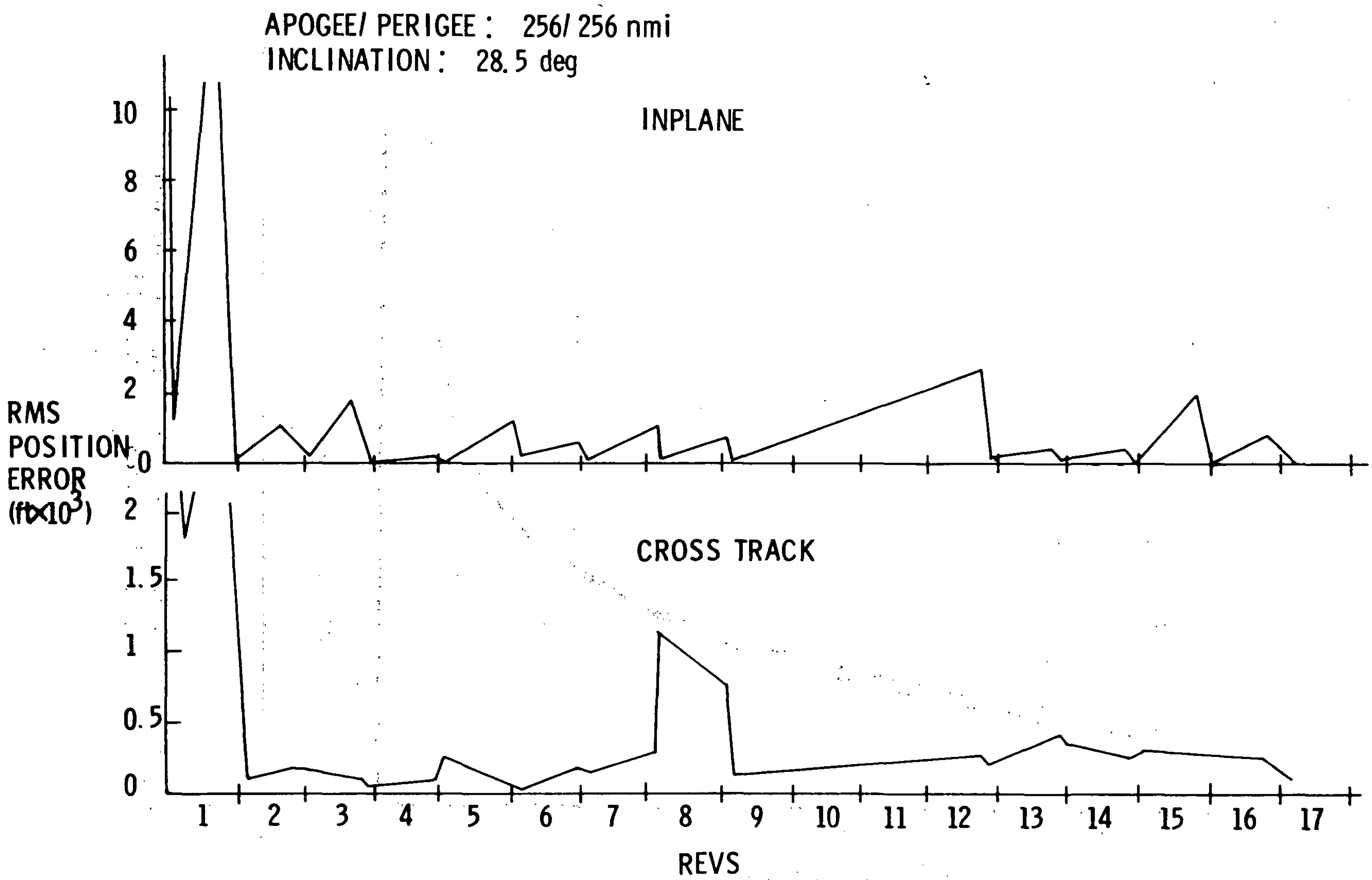


Figure 3-4. Ground Transponder Navigation Performance (Reproduced from Reference 20)

The results represent tracking on the transponder system defined in Figure 3-1 for a period of 16 orbital revolutions (one earth rotation). The position error during the tracking of any transponder reduces to less than 500 ft (1σ). The maximum rate of position error growth between transponders is approximately 2500 ft/rev. An anomaly is noted in the crosstrack performance in revolution 8. This is due to the orbit passing almost directly over a transponder which can cause a degradation in crosstrack performance as the result of unmodeled systematic errors in the measurement data (Reference 19).

Figure 3-5 contains the navigation errors for a 137 nautical mile circular orbit at 90 degree inclination. The results here exhibit a larger error growth between transponders (approximately 4000 ft/rev maximum) than the error in Figure 3-4 due to the reduced length of tracking time over the transponders at the lower orbital altitude.

Studies of the convergence of navigation errors during initial tracking intervals have shown that data from two transponders is required to achieve steady state performance. Data from a single transponder provides an accurate local position estimate but cannot resolve the velocity uncertainty to a level that produces an acceptable error growth. The rate of error growth following initial tracking on a single transponder is quite sensitive to differences in the real-world and filter apriori covariances as shown by the curves in Figure 3-6. The real-world initial covariance was initialized with strong correlations within the inplane and crosstrack state elements. In Case #1, the navigation filter covariance was initialized with the diagonal elements of the real-world covariance. In Case #2, the filter and real-world covariances were set equal. The significant error growth after transponder #1 (Case #1) emphasizes the inability to adequately resolve the inplane errors from tracking a single transponder. After tracking the second transponder, the error behavior for the two cases becomes essentially identical, the navigation filter having achieved a steady state performance independent of the initial errors. The initial error growth exhibited in Case #1 can be reduced considerably by correlating the inplane state elements in the filter apriori covariance. It is however, still necessary to track a minimum of two transponders to reduce large initial errors to the desired steady state performance level.

The navigation results discussed here were obtained with a filter mechanization that included no estimation of systematic measurement errors. The error model previously presented contains no error source that significantly degrades the navigation accuracy.

A summary of the performance is:

(1) The distribution of seven transponders shown in Figure 3-1 provides adequate coverage for all space station orbits except those with near equatorial inclination.

(2) The steady state navigation performance maintains the position error to within approximately 4000 ft (1σ). For the majority of orbits, the position error is less than 1000 ft (1σ).

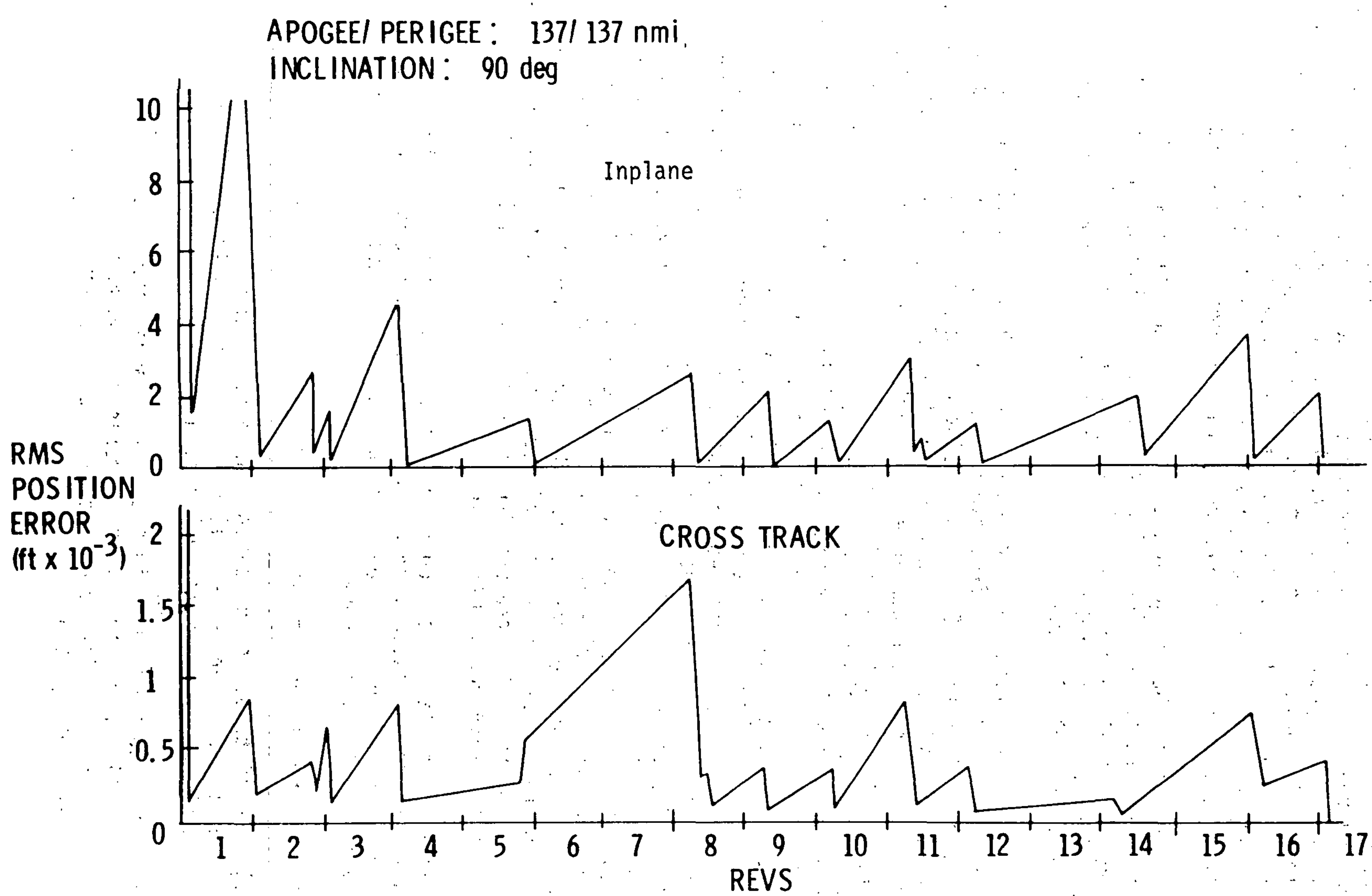


Figure 3-5. Ground Transponder Navigation Performance (Reproduced from Reference 20)

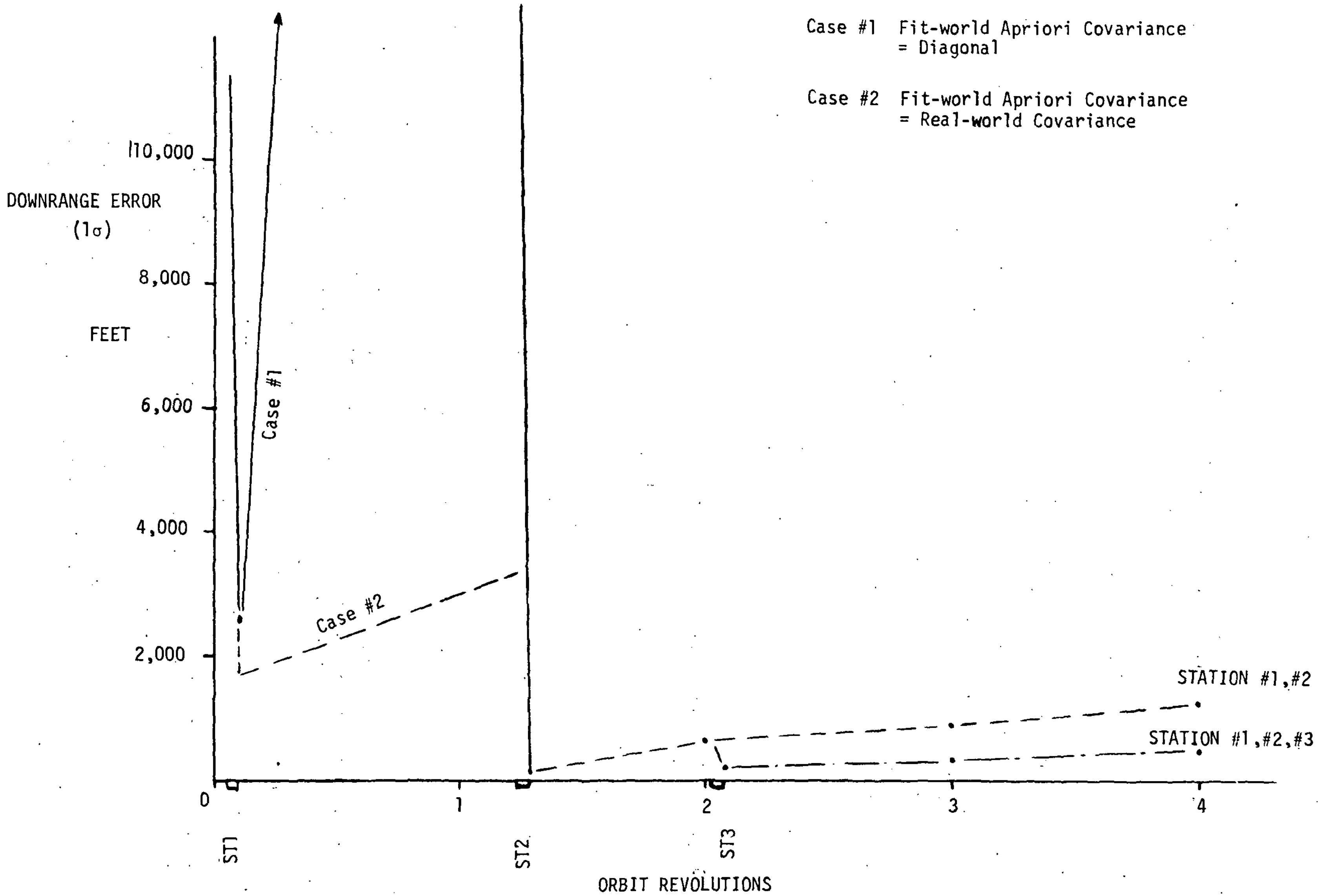


Figure 3-6. Ground Transponder Navigation Error Convergence Characteristics

(3) Tracking data from two transponders is required to reduce large initial state errors and achieve steady state navigation accuracy.

(4) Tracking data at elevation angles below 5 degrees is highly undesirable due to the potentially large error in the refraction correction model.

Horizon Sensor Navigation System

The star/horizon orbital navigation system consists of a set of horizon sensors that measure the angle to the earth's horizon with respect to vehicle fixed axes and a stellar inertial referenced system that determines the orientation of the vehicle axes in inertial coordinates. The measurements used for navigation are derived by combining the horizon sensor data with the onboard estimate of vehicle inertial attitude to form a "star"/horizon angle measurement where the inertial reference is used to form a fictitious star line-of-sight. The geometry of the star/horizon measurement is shown in Figure 3-7. The horizon measurement angle is sensitive to navigation errors in the measurement plane defined by the horizon line-of-sight and the vehicle position vector. The horizon sensor system is mechanized to perform measurements at several sighting azimuths relative to the orbital plane to provide inplane and crosstrack navigation updates.

The stellar referenced coordinate system could be determined through the use of continuous monitoring of the angles to two stars. This would required two gimballed star trackers with fields of view arranged so that two stars at a sufficient angular separation could be tracked continuously. Another mechanization approach is to use a gyro system for short term angular reference and periodically update it with stellar observations to maintain the desired long term accuracy. In this case, the gyro reference could be provided in the form of a gimballed stabilized platform or a strapdown gyro reference package.

The horizon sensor operation consists of scanning the earth's horizon with a narrow band radiation detector. The intensity of the sensed radiation for a given frequency band is a function of the altitude above the earth's surface as shown in Figure 3-8. The reference horizon is termed a locator and is determined as some function of the horizon profile curve. Two of the commonly used locators are 1) fixed radiance level, and 2) integral of the radiance to a selected point. The horizon profile at any specific location on the earth is highly variable, being a function of cloud cover, latitude, season, and local atmospheric conditions. The seasonal variation is greatest at higher latitudes, reducing to zero at the equator. The horizon defined by the locator will also vary with the altitude of the spacecraft. The variation in observed altitude with spacecraft altitude for a fixed radiance locator is illustrated in Figure 3-9.

Sensors containing IR (infrared) radiation detectors can track the horizon throughout the orbit while those containing UV (ultraviolet) detectors are restricted to tracking the sunlit portion of the earth. On

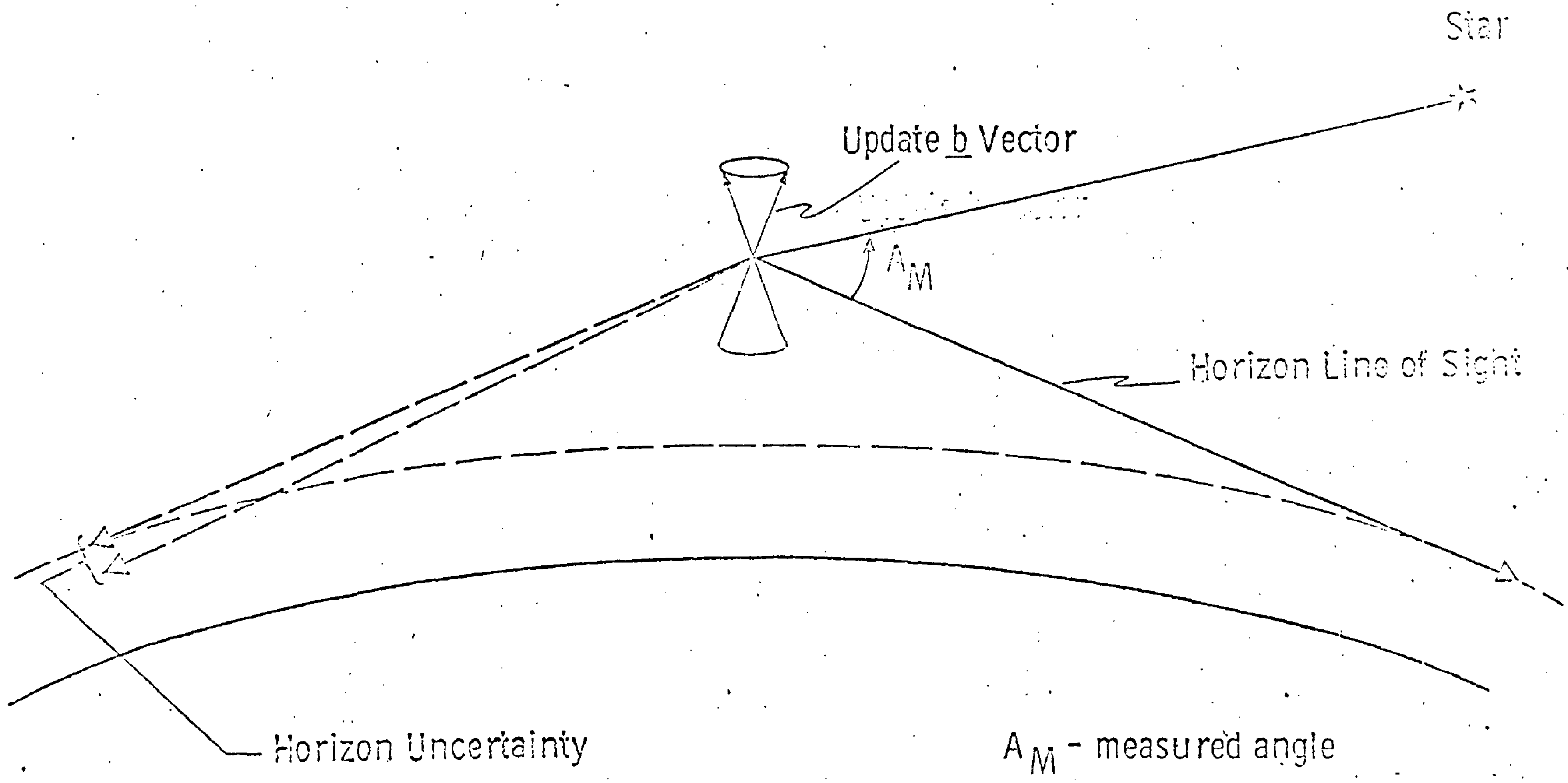


Figure 3-7. Star/Horizon Measurement Gravity

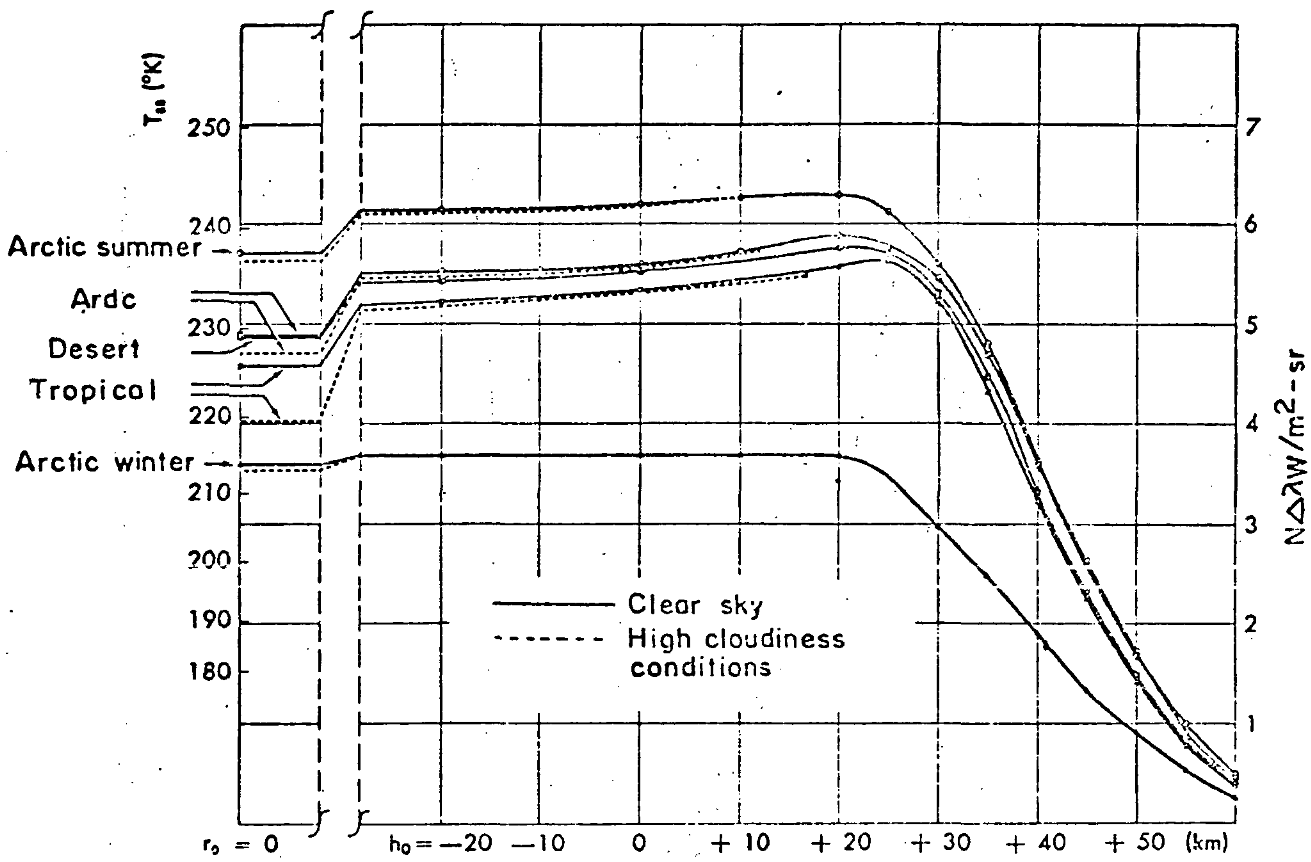


Figure 3-8. Curves of Radiance in the 15 Micron CO_2 Band Versus Altitude for Five Model Atmospheres (Reproduced from Reference 21)

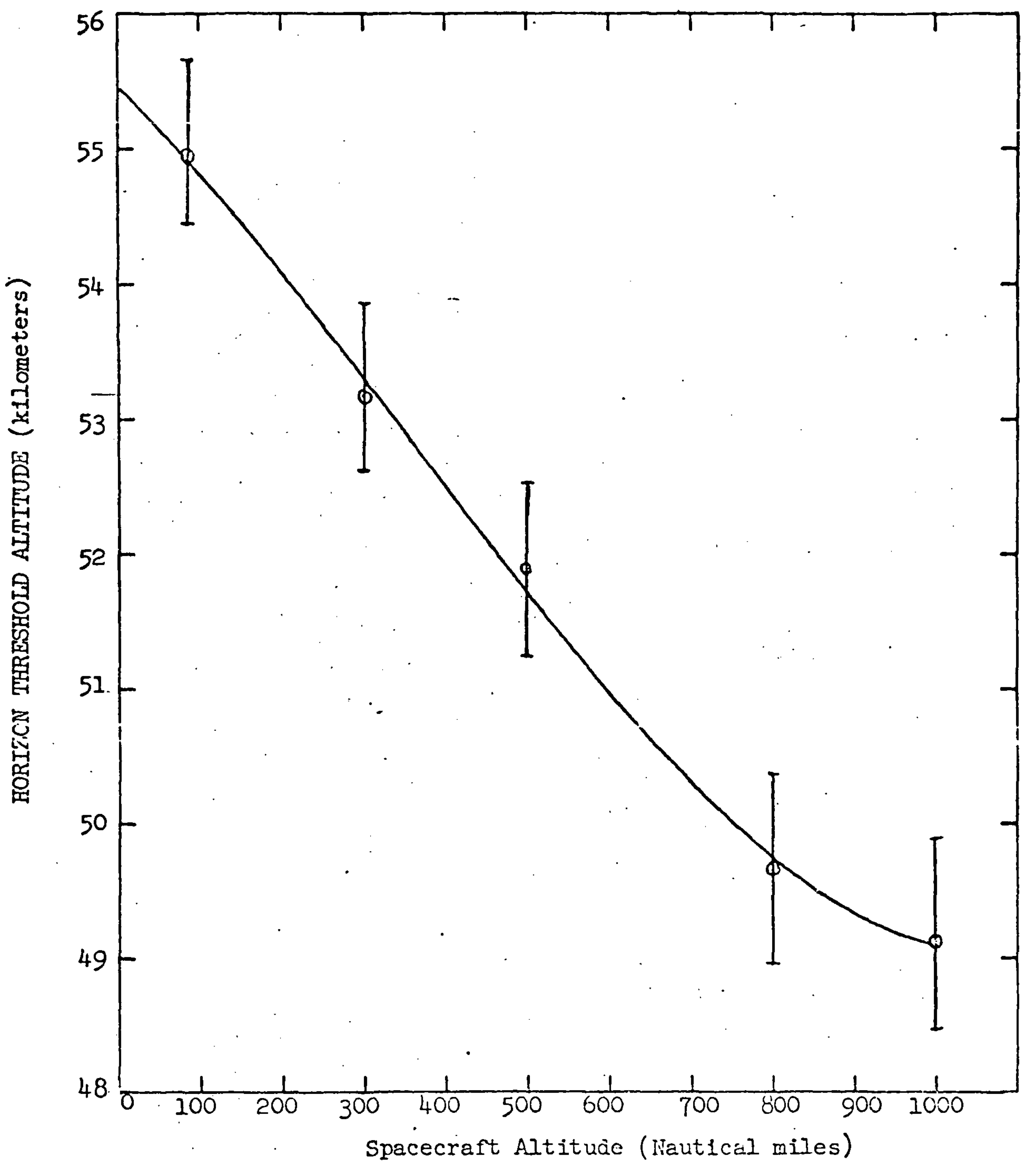


Figure 3-9. Variation of the observed altitude of the infrared horizon corresponding to a radiance of 0.6 watts/meter^2-steradian and due to field of view dimensions, response time, and spacecraft altitude. A scan rate of 1.0 degree per second, and a response time of 250 milliseconds were used in the calculation. Effects of a peak-to-peak signal-to-noise ratio of 10 to 1 are shown with vertical bars. (Reproduced from Reference 22)

each orbit, the preferred measurement zone is between ± 40 degrees latitude, the region where the variations in horizon altitude can be modeled to a reasonable accuracy.

System Error Model

The error in the "star"/horizon angle measurement can be attributed to the following error sources:

(1) Misalinement of the inertial attitude reference;

(2) Horizon sensor error;

(3) Unmodeled variations in the horizon locator altitude.

The misalinement in the attitude reference is caused by star tracker sighting errors during attitude updates plus drift due to gyro output errors. The 1σ values for random and bias attitude errors about any vehicle axes are given in Table 3-2. Table 3-2 also contains a preliminary error budget for an IR and a UV system for low earth orbit application.

The dominant error in the horizon sensor measurement is the uncertainty in the horizon altitude. Onboard mathematical models can be used to compensate for the known variations as a function of latitude, season, etc. The residual altitude error after compensation, resulting from unmodeled atmospheric effects, is commonly modeled as a random process with both a time and spatial correlation. The variation with time has been observed to have a time constant in excess of one day; consequently, the model for the horizon altitude uncertainty can be simplified to account for only spatial correlations as represented in the following autocorrelation function form:

$$\phi(d) = \sigma_h^2 \; e^{-\frac{|d|}{\delta}}$$

where

$\phi(d)$ is the autocorrelation for the spatially correlated horizon altitude error

d is the great circle distance between the horizon points

σ_h^2 is the variance for the horizon altitude uncertainty

δ is the correlation distance constant

For the IR horizon sensor system, the horizon altitude error model values consist of a standard deviation (σ_h) of 3000 feet with a correlation distance (δ) of 2500 nautical miles. The UV horizon profile has not been analyzed nearly as much as the IR profile. Accuracies currently being used for the UV horizon after compensation for known functional variations are 3000 ft (1σ) for the bias term and 3000 ft (1σ) for the noise.

Table 3-2. Horizon Sensor System Error Model

		Random (sec 1σ)	Bias (sec 1σ)
<u>IR SENSOR ERRORS</u>			
Output Noise		75	0
Mounting		--	8
Visor Bias (Mirror Assembly)		--	15
Null Position Detector & Alignment		--	18
Output Quantization		20	
	RSS	80	25
<u>UV SENSOR ERRORS</u>			
Linearity		12	0
Output Noise		30	0
Optics (distortion, drift, truncation)		12	10
Null Position		--	25
Gimbal Readout		20	15
Gimbal Alignment		--	25
	RSS	40	40
<u>ATTITUDE REFERENCE ERRORS</u>			
Tracker (gimbal readout, alignment, etc.)		--	35
Gyros & Software		10	20
	RSS	10	40

Navigation Performance

A basic conclusion of horizon sensor system navigation performance studies is that the horizon altitude uncertainty is the major error source and that significant improvement in the navigation accuracy can be achieved by estimating the altitude bias in the onboard filter. The 1σ navigation accuracy for an IR horizon sensor system operating in a 240 nautical mile orbit at 55 degree inclination is presented in Figure 3-10 (Reference 23). The navigation schedule consisted of inplane and crosstrack horizon sensor measurements taken at 5 minute intervals for the entire orbit, with the navigation filter mechanized to estimate the horizon altitude bias. The error curves show the navigation performance to converge to a steady state position accuracy of 4400 ft (1σ) in approximately one and a half orbits (140 minutes) of tracking. The 1σ position error components in steady state are 3800 feet downrange, 1100 feet radial, and 1800 feet corssrange. For a UV horizon sensor system, similar steady state accuracy is expected although the period of convergence would be greater due to the lack of measurements during one half of each orbit.

A summary of the performance is:

(1) The dominant error source is the residual horizon altitude error after onboard compensation for known variations. A significant improvement in navigation accuracy is achieved by estimating the horizon bias in the onboard filter.

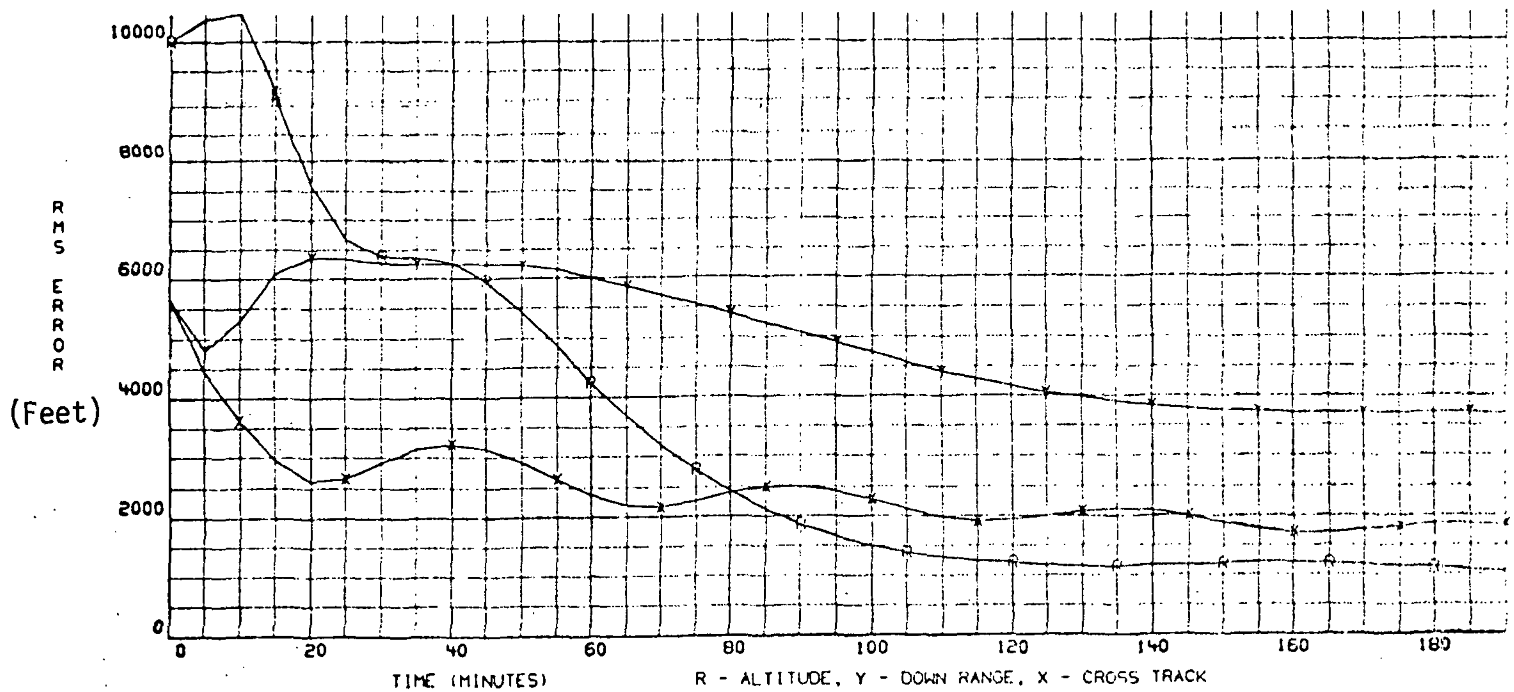
(2) Steady state navigation accuracy is achieved in approximately one and a half orbits of tracking for an IR system.

(3) The steady state navigation performance has a total position uncertainty of 4400 ft (1σ).

TDRS (Tracking and Data Relay Satellite System)

The TDRS navigation system is composed of a configuration of one to three satellites in synchronous orbit with inclinations less than 5 degrees. These satellites act as relays from the ground stations to the user spacecraft. As such, the basic measurements are (1) the total range from the ground station, and (2) the sum of the range rate between the ground and the TDRS and the range rate between the TDRS and the user spacecraft. Because the TDRS is essentially stationary with respect to the ground stations, the range rate measurement is essentially the range rate between the TDRS and the user spacecraft. The capabilities and characteristics of the TDRS are discussed in References 5 and 24. The field of view of the TDRS is approximately 13° half cone angle around the vector to the center of the earth. This provides coverage for spacecraft out to about 1000 miles altitude.

An error budget for this system is given in Table 3-3. Two effects which are not included in this table are multipath and refraction. The refraction effect will be almost constant because the elevation angle from the ground station to the TDRS will be constant. This should allow almost complete compensation with the software. From published documents, it appears that



Navigation accuracy for an infrared sensor system
Operating on a 240 Nautical Mile Orbit

Figure 3-10. Star/Horizon System Orbit Navigation Performance (Curve Reproduced from Reference 23)

Table 3-3. TDRS Error Model (1σ)

RANGE			
Noise			
Reference Oscillator			0.03 ft
Quantization			14.1 ft
Phase Lock Loop Error			138.2 ft
		RSS	140 ft
Bias			
Reference Oscillator	Scale Factor		10^{-5} PPM
Speed of Light	Scale Factor		0.5 PPM
Effective Additive Bias			90. ft
RANGE RATE (S-BAND)			
Noise			
Reference Oscillator			0.05 ft/sec
Quantization (1 second)			0.03 ft/sec
Phase Lock Loop Error			0.01
		RSS	.06 ft/sec
Bias			
Reference Oscillator	Scale Factor		10^{-5} PPM
Speed of Light	Scale Factor		0.5 PPM

Location Error

Radial	50 ft.
Tangential	900 ft.
Out-of-Plane	400 ft.

multipath has not been analyzed in sufficient depth to develop an error contribution. However, it is recognized as a problem, particularly for a low orbit user with an omni antenna. The effect on the range measurement of the reference oscillator long term instability and the speed of light uncertainty is as a scaling error. This error will appear essentially as a constant additive bias for low earth orbit users because the total range will be almost constant during all tracking intervals.

Navigation Performance

The navigation capability with a TDRS was analyzed in Reference 25 considering both the accuracy of determining the TDRS orbit from ground tracking data and the accuracy of determining the user orbit from the TDRS tracking data. The analysis of the TDRS orbit determination accuracy used the MSFN tracking data uncertainties (Table 3-7) and 24 hours of tracking data at a frequency of two measurements per hour. The essential results of this analysis are:

(1) When the TDRS is in a 5° inclined orbit, one ground station with range measurements can provide an RMS accuracy of 2000 feet with propagation errors of less than 1000 feet per revolution (24 hour period). The addition of range rate measurements and/or a second station can improve the initial uncertainty but degrade the propagation characteristics to about 1000 feet per revolution.

(2) When the TDRS is in an equatorial orbit, range or range and range rate from one ground station cannot improve upon the apriori downrange uncertainty. The addition of angle measurements can provide an RMS accuracy of 4000 feet. Two ground stations with range measurements can provide an RMS accuracy of 1000-3000 feet per revolution. The addition of range rate will provide 1000 feet initially and 1000 feet per revolution propagation error.

One very important output of this analysis is that the period of the TDRS is well determined and therefore the growth rate of the TDRS state vector error is small.

A TDRS configuration with two equatorial TDRS located at 28 degrees West longitude and 164 degrees West longitude was considered in Reference 25 to analyze user orbit determination accuracy in a 55° inclined circular orbit. The measurement accuracies were taken to be:

Measurement	Noise (1σ)	Bias (1σ)
Range	30 ft	30 ft
Range Rate	.032 ft/sec	.028 ft/sec

The TDRS uncertainties were taken to be as previously discussed with two ground stations measuring range and range rate. This error model is slightly optimistic when compared to the error budget of Table 3-3. The results of the analysis indicates that the user navigation accuracy after two orbits (180 minutes) of data varies from 200 feet to 1200 feet RMS error depending upon the measurement schedule, but with at least 5 minutes of data every 45 minutes.

The analysis of Reference 26 considered a three TDRS configuration and a one TDRS configuration with only range rate measurements. The measurement error was considered as 0.33 and 0.67 feet per second noise (1σ) with and without a bias of the same value. These results indicate that a single TDRS can provide an RMS accuracy of 1000 feet with range rate measurements taken every 5 minutes for 4 revolutions (Figure 3-11). The three TDRS configuration provides performance superior to the one TDRS, particularly in convergence time. One revolution of measurements from the three TDRS is sufficient except for low inclinations. For low inclinations (<5°) the out-of-plane error converges slowly and at zero degrees inclination the out-of-plane error appears to be unobservable.

The performance of a two TDRS configuration appears to be in between the results of References 25 and 26 whose measurement error models bracket the error budget of Table 3-3. The measurement schedule appears to be the most important factor in the navigation accuracy except when the inclination is near zero degrees. The performance of the system can be described by:

Convergence time - 1 revolution

RSM Accuracy - 500 - 1000 feet

MSFN (Manned Spaceflight Network)

The MSFN is a system of unified S-band tracking, telemetry, and command stations and other sites with C-band tracking, VHF telemetry and air/ground voice capabilities established, maintained, and operated by the Goddard Space Flight Center (GSFC) Manned Flight Support Directorate. In addition, there are sites of the Eastern Test Range (ETR) supporting these missions in the launch phase which are linked with the NASA Communications Network (NASCON). The primary purpose of the MSFN is to support manned space missions, both earth orbital and lunar, although it is called upon to support other progams.

- Unified S-Band Systems (USB) - The USB tracking systems determine the position of a spacecraft by measuring X and Y angles, range, and range rate of the vehicle with respect to the radar. The stations are equipped with either 85-ft or 30-ft diameter antennas. There are always two 85-ft stations located close together; one being a MSFN, the other, a Deep Space Network (DSN) station. The DSN stations are directed by the Jet Propulsion Laboratory. This network is used for lunar and deep space exploration programs and is also used to support Apollo missions. Discussions of the DSN will be included with those of the MSFN since the sites are co-located and because of the similarity of their

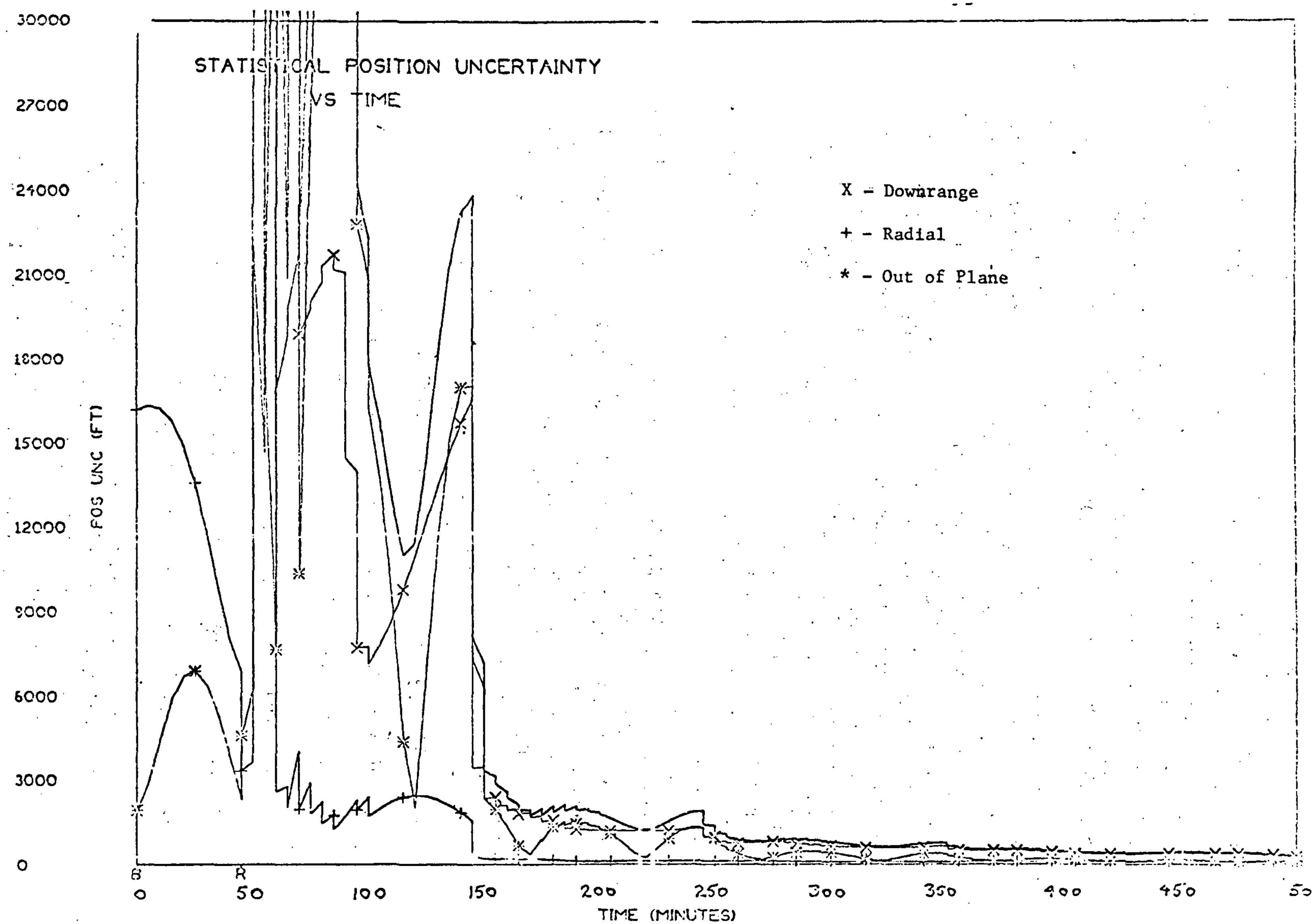


Figure 3-11. Navigation Accuracy from a Single Synchronous Satellite
(Reproduced from Reference 26)

equipment. The 85-ft antennas associated with either station can be utilized to simultaneously track two spacecraft. Each 85-ft "dual site" employs two frequency separated, receive-transmit links to permit simultaneous tracking and communication with two spacecraft provided both craft are within the antenna beam. Accurate angle data are obtained only for the spacecraft onto which the antenna is locked. As indicated in Table 3-4, many of the 30-ft USB sites also have dual capability.

- Satellite Tracking and Data Acquisition Network (STADAN) - The primary purpose of the STADAN is to receive data from scientific satellites and to produce tracking information for orbit computation. The station location and equipment for each of the STADAN sites is shown in Table 3-5. Most of the equipment in the STADAN has been designed for use by many programs, with emphasis on quick adaptability to the differing requirements of several simultaneously orbiting spacecraft. Most programs do not require data from all STADAN stations, so the specific capabilities of each station have been tailored to differing levels of performance. The result is that no two stations are identical in terms of either equipment types or total data capacity.

  Operation of the STADAN is centered at GSFC. All stations are connected to GSFC through teletype (TTY) and voice (SCAMA) lines. Fairbanks and Rosman also have wide-band microwave links to GSFC for real time data transmissions. Certain "quick-look" telemetry data is sent to GSFC over TTY lines in near real time, but most telemetry data is recorded and sent to GSFC by mail. Tracking data is sent by TTY in near real time.

  The STADAN consists of five major systems:

  - 85-foot Data Acquisition Facility (DAF)
  - 40-foot DAF
  - VHF Telemetry System
  - Goddard Range and Range Rate System (GRARR)
  - Minitrack Tracking System

  The most obvious distinction between these systems is the type of antenna used. The first three systems all use identical or very similar receivers, transmitters, and data handling equipment. Only the GRARR and Minitrack Systems are completely unique. A particular station may have any combination of the above systems, and the

Table 3-4. MSFN Station Locations

Station Name	Designation	Description	Latitude (deg North)	Longitude (deg East)	Height (meters)
Merritt Island, Florida	MIL - 3	*USB - 30'	28.50827	279.30619	-18.
	MLA - T	TPQ - 18	28.42486	279.33520	-16.
Greenbelt, Maryland	ETC - 3	USB - 30'	38.99848	283.15683	55.
Bermuda	BDA - 3	USB - 30'	32.35125	295.34150	- 2.
	BDA - F	FPS - 16	32.34807	295.34588	- 5.
	BDA - Q	FPQ - 6	32.34793	295.34594	- 4.
Canary Island	CYI - 3	USB - 30'	27.76454	344.36615	173.
Ascencion Island	ACN - 3	*USB - 30'	- 7.95479	345.67269	522.
Carnarvon, Australia	CRO - 3	*USB - 30'	-24.90658	113.72580	- 15.
	CRO - Q	FPQ - 6	-24.89639	113.71763	- 11.
Guam	GWM - 3	*USB - 30'	13.31058	144.73634	143.
Kauai, Hawaii	HAW - 3	*USB - 30'	22.12631	200.33480	1135.
	HAW - F	FPS - 16	22.12350	200.33441	1139.
Guaymas, Mexico	GYM - 3	USB - 30'	27.96321	249.27875	- 22.
Corpus Christi, Texas	TEX - 3	USB - 30'	27.65375	262.62113	- 20.
Madrid, Spain	MAD - 8	*USB - 85'	40.45498	355.83155	778.
	MAD - W (DSN)	*USB - 85'	40.42829	355.75101	770.
Honeysuckle Creek (Canberra, Australia)	HSK - 8	*USB - 85'	-35.58349	148.97783	1145.
	HSK - W (DSN)	*USB - 85'	-35.40099	148.98130	670.
Goldstone, California	GDS - 8	*USB - 85'	35.34159	243.12640	907.
	GDS - W (DSN)	*USB - 85'	35.38957	243.15063	971.
Vandenberg, AFB California	CAL - T	TPQ - 18	34.66590	239.44296	45.

* Designate "dual" Unified S-Band capability

Table 3-5. STADAN Station Locations

Station Name	Designation	Description	Latitude (deg North)	Longitude (deg East)	Height (meters)
Fort Meyers, Florida	FTMYRS	Minitrack	26.54827	278.13461	9.
Quito, Ecuador	QUITOE	Minitrack	- 0.62238	281.42101	3578.
		40' DAF	- 0.62305	281.41932	3580.
Santiago, Chile	SNTAGO	Minitrack	-33.14895	289.33136	681.
		GRARR - S	-33.15048	289.33326	695.
		GRARR - VHF	-33.15117	289.33326	695.
		40' DAF	-33.15090	289.33135	691.
Lima, Peru	LIMAPU	Mintrack	-11.77635	282.84976	34.
Goldstone, California	GLDSTN	Minitrack	35.33016	243.10024	921.
		40' DAF	35.33179	243.11274	933.
Tananarive, Madagascar	MADGAR	Minitrack	-19.00700	47.30013	1361.
		GRARR - S	-19.01926	47.30349	1409.
		GRARR - VHF	-19.01994	47.30349	1409.
		40' DAF	-19.00956	47.30157	1395.
St. Johns, Newfoundland	NEWFLD	Minitrack	47.74137	307.27964	112.
Fairbanks, Alaska	ALASKA	Minitrack	64.97740	212.47800	330.
		GRARR - S	64.97217	212.48761	371.
		GRARR - VHF	64.97170	212.48920	371.
		85' DAF	64.97684	212.48293	333.
		40' DAF	64.97662	212.47970	323.
Winkfield, England	WNKFLD	Minitrack	51.44595	359.30377	87.
Rosman, North Carolina	ROSMAN	GRARR - S	35.19603	277.12414	876.
		GRARR - VHF	35.19493	277.12414	876.
		85' DAF	35.20020	277.12812	895.
		85' DAF	35.19898	277.12445	890.
Johannesburg, South Africa	JOBURG	Minitrack	-25.88361	27.70791	1565.
		40' DAF			
Carnarvon, Australia	CARVON	GRARR - S	-24.90413	113.71529	51.
		GRARR - VHF	-24.90523	113.71529	51.
Orroral, Australia	ORORAL	Minitrack	-35.62709	148.95298	947.
		85' DAF			

specific configuration of the system will vary depending on the requirements placed on the station.

The 85-ft and 40-ft DAF are instrumented for telemetry reception at 136 MHz, 400 MHz, and 1700 MHz, except the Goldstone 40-ft dish which is exclusively instrumented for use by the Advanced Technology Satellite (ATS) project. These systems have VHF capability through an auxiliary antenna.

The GRARR system is better adapted to support future manned missions than the 85-ft and 40-ft DAF systems. A modification program now underway will modify the GRARR frequencies to make it compatible with Apollo and the Air Force SGLS system. With some additional modifications, the GRARR sites could be capable of operating as a Unified S-Band system.

Use of the Minitrack system for accurate tracking and orbit computation would require the use of a 136 MHz beacon onboard the spacecraft. Minitrack has the capability of determining an accurate orbit in a period of a few hours.

- Combination MSFN and STADAN Network - A combination of the MSFN and STADAN networks has been considered which will result in a more flexible and economical system (Reference 27). The stations comprising this network (Table 3-6) and the resulting visibility characteristics were taken from Reference 28 and are shown as Figures 3-12, 3-13, and 3-14 for a 235 nautical mile circular orbit. The summary for the coverage is illustrated in Figure 3-15. As seen in this figure, coverage by two or more stations per revolution is assured for the 30 degree inclined orbit. This level of coverage is provided to the 50° and 90° inclination orbits approximately 90% of the time. The vehicle was viewed by no stations on one orbit for the 50° inclined orbit.

System Error Model

Error model values for the MSFN tracking network stations are presented in Table 3-7.

Navigation Performance

The navigation performance obtained from MSFN tracking data is characterized by small state vector uncertainties during tracking intervals followed by error growth in the inplane components during intervals of non-coverage. Tracking data from a minimum of two stations is required to reduce large initial state errors to a steady state performance level. Steady state position uncertainties are on the order of 300 to 900 feet (1σ) immediately following station tracking with an error growth rate less than 1000 feet (1σ) per revolution.

Table 3-6. Stations of the Network Resulting from Combining STADAN and MSFN

Station Name	Designation	Description	Latitude (deg North)	Longitude (deg East)	Height (meters)
Madrid, Spain	MAD - 8	*USB - 85'	40.45498	355.83155	778.
	MAD - W (DSN)	*USB - 85'	40.42829	355.75101	770.
Honeysuckle Creek (Canberra, Australia)	HSK - 8	*USB - 85'	-35.58349	148.97783	1145.
	HSK - W (DSN)	*USB - 85'	-35.40099	148.98130	670.
Goldstone, California	GDS - 8	*USB - 85'	35.34159	243.12640	907.
	GDS - W (DSN)	*USB - 85'	35.38957	243.15063	971.
Guam	GWM - 3	*USB - 30'	13.31058	144.73634	143.
Ascension Island	ACN - 3	*USB - 30'	- 7.95479	345.67269	522.
Merritt Island, Florida	MIL - 3	*USB - 30'	28.50827	279.30619	-18.
	MLA - T	TPW - 18	28.42486	279.33520	-16.
Corpus Christi, Texas	TEX - 3	USB - 30'	27.65375	262.62113	-20.
Kauai, Hawaii	HAW - 3	*USB - 30'	22.12631	200.33480	1135.
	HAW - F	FPS - 16	22.12350	200.33441	1139.
Bermuda	BDA - 3	USB - 30'	32.35125	295.34150	- 2.
	BDA - F	FPS - 16	32.34807	295.34588	- 5.
	BDA - Q	FPQ - 6	32.34793	295.34594	- 4.
Rosman, North Carolina	ROSMAN	GRARR - S	35.19603	277.12414	876.
		GRARR - VHF	35.19493	277.12414	876.
		85' DAF	35.20020	277.12812	895.
		85' DAF	35.19898	277.12445	890.
Johannesburg, South Africa	JOBURG	Minitrack	-25.88361	27.70791	1565.
		40' DAF			
Quito, Ecuador	QUITOE	Minitrack	- 0.62238	281.42101	3578.
		40' DAF	- 0.62305	281.41932	3580.
Fairbanks, Alaska	ALASKA	Minitrack	64.97740	212.47800	330.
		GRARR - S	64.97217	212.48761	371.
		GRARR - VHF	64.97170	212.48920	371.
		85' DAF	64.97684	212.48293	333.
		40' DAF	64.97662	212.47970	323.

Notes:

3 letter designators indicate original MSFN stations

6 letter designators indicate original STADAN stations

USB - Unified S-Band

GRARR - Goddard Range and Range Rate System

DAF - Goddard Data Acquisition Facility

In addition, the network includes one ship and one land transportable (Europe) tracker

Table 3-7. MSFN Measurement Accuracy

Measurement	Noise (1σ)	Bias (1σ)
Range	30 ft	60 ft
Range Rate	.002 ft/sec	.03 ft/sec
Angles	0.15 MR	1.6 MR

STATION LOCATION UNCERTAINTIES (1σ)

Latitude	Longitude	Altitude
1.0 $\widehat{\text{sec}}$	0.8 $\widehat{\text{sec}}$	140 ft

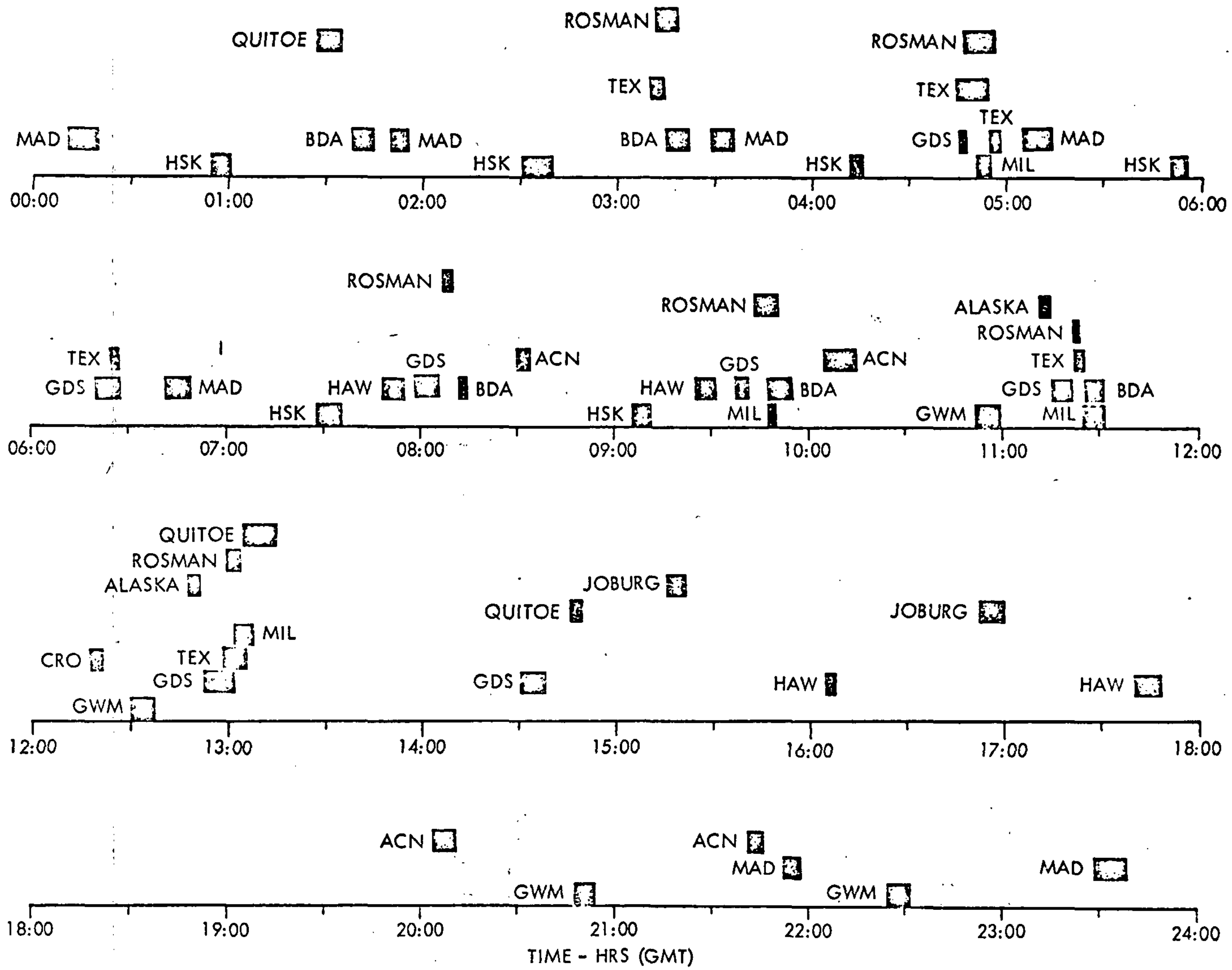


Figure 3-12. Network Visibility of 50° Inclination, 235 n.mi. Circular Orbit

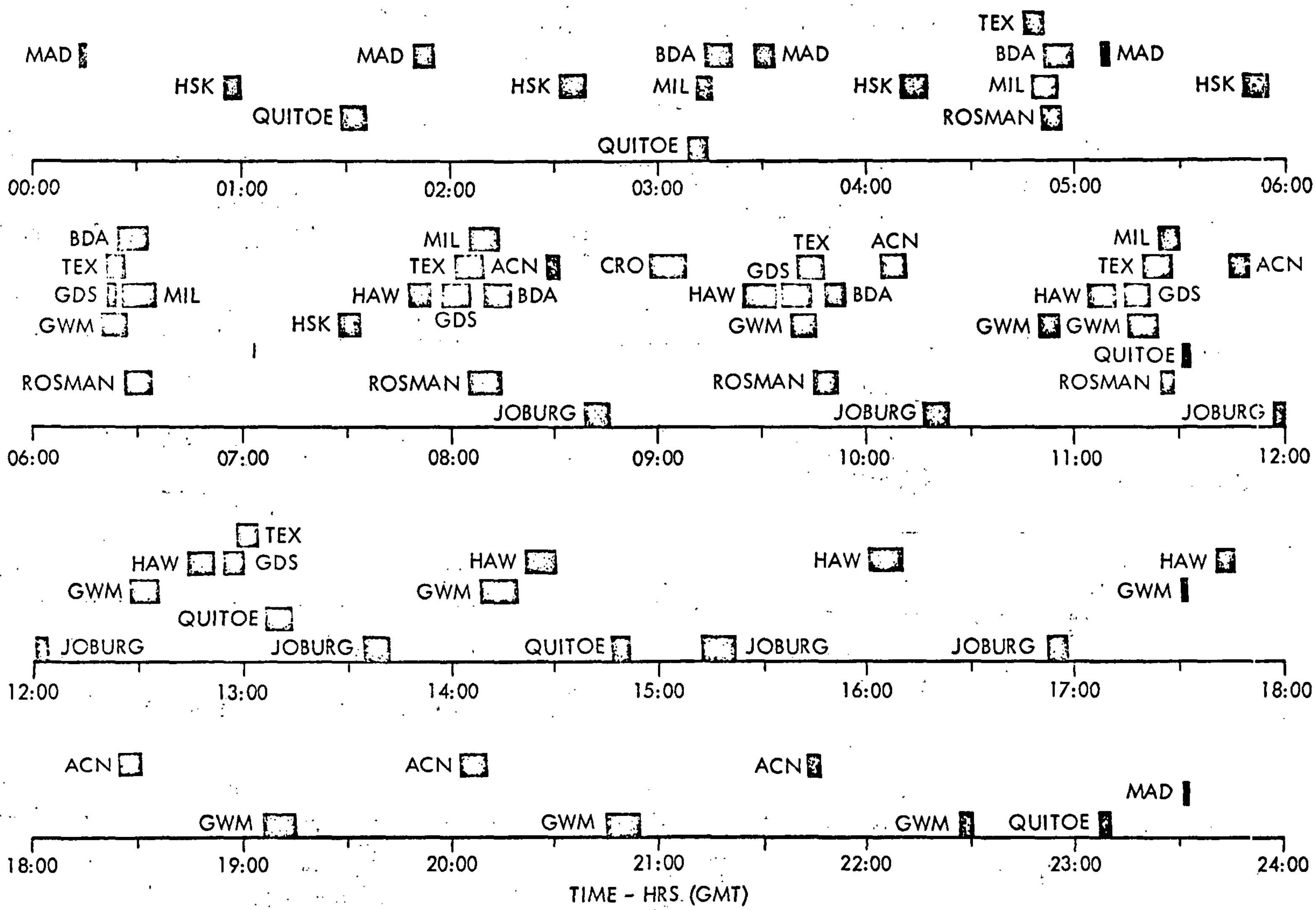


Figure 3-13. Network Visibility of 30° Inclination, 235 n.mi. Circular Orbit

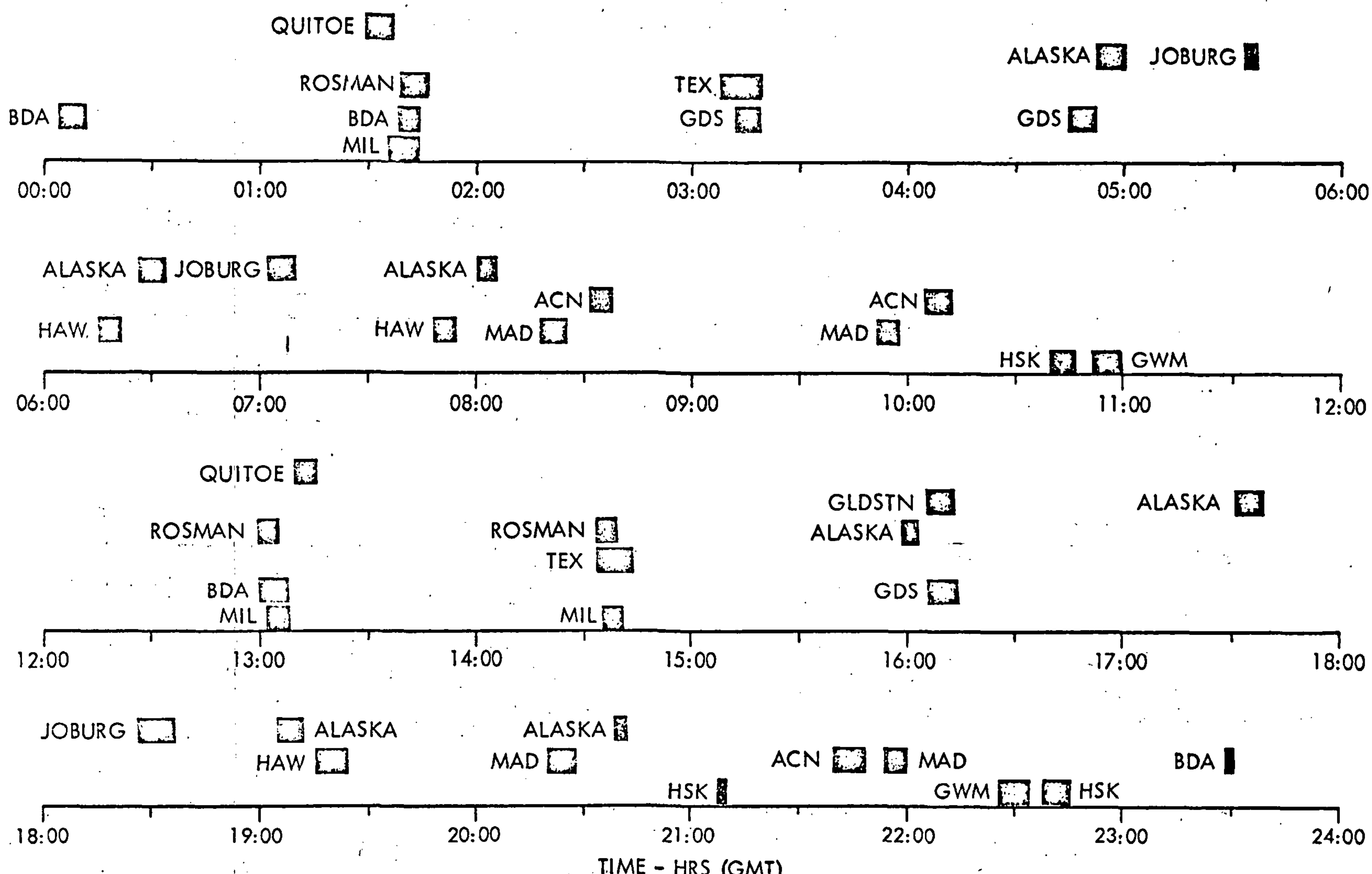


Figure 3-14. Network Visibility of 90° Inclination, 235 n.mi. Circular Orbit

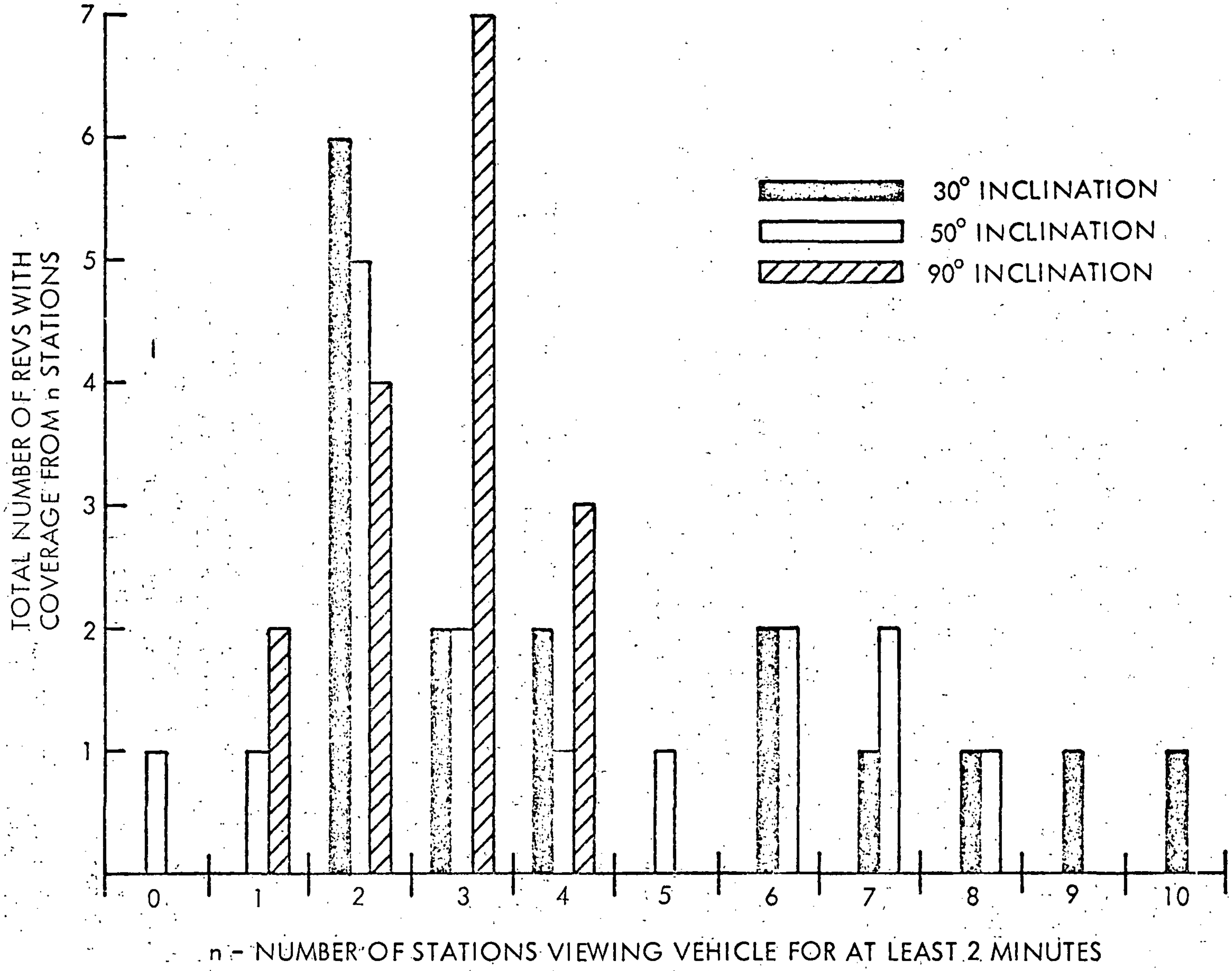


Figure 3-15. Histogram of Coverage Levels

Unknown Landmark Tracker Navigation System

The system considered consists of an unknown landmark tracker that measures the line-of-sight to a landmass feature on the earth's surface and an inertial attitude reference. The unknown landmark tracker consists of a specific implementation of the generic class of electro-optical correlation trackers. Specifically, it is a two-dimensional intensity correlation tracker implemented in an analog/digital format employing a nulling track mode and operating in the visible portion of the spectrum. Trackers of this and similar types have been studied, designed, and implemented for several applications for various military agencies over the past 15 years.

During normal operation, the tracker servos point the optical line-of-sight in an arbitrary direction with respect to body axes and a reference image representing an area on the surface of the earth is stored. Subsequently, the tracker performs a two-axis correlation of its current image with the reference image, obtaining error signals which it uses to drive the gimbal servos to keep the optical axis pointed at the same landmark.

One mechanization approach for the attitude reference is to use a gyro system for short term angular reference and periodically update it with stellar observations to maintain the desired long term accuracy. The gyro reference would most likely be provided in the form of a strapdown gyro package. The use of the landmark tracker for star tracking during periods of landmark unavailability is a possible mechanization provided the field-of-view maneuverability is adequate.

System Error Model

The navigation performance of this system is dependent upon the measurement accuracies, the availability of sunlit landmass targets, and the terrain uncertainties of the targets. A preliminary error budget developed for this system for low earth orbit application is given in Table 3-8.

Navigation Performance

The navigation performance of the unknown landmark tracker system is characterized by a steady state position uncertainty of about 2000 ft (1σ) during tracking of landmasses during the daylight portion of the orbit followed by inplane error growth of 1000 to 2000 ft (1σ) per revolution on the nightside pass. The state error convergence characteristics require tracking on at least 5 to 6 landmarks over a one revolution interval to obtain steady state performance. A significant limiting factor on the realizable accuracy is the availability of sunlit landmass targets, which can be restricted by cloud cover. A viable option is to include a horizon sensor system with the unknown landmark tracker to insure adequate convergence.

Typical navigation accuracy for a landmark tracker system augmented with horizon sensor data is presented in Figure 3-16.

Table 3-8. Unknown Landmark Tracking
System Error Budget

Unknown Landmark Tracking

Error Source		Random ($\widehat{\text{sec}}$, 1σ)	Bias ($\widehat{\text{sec}}$, 1σ)
Atmospheric refraction, turbulence, dispersion		1	
Scene Truncation		1	
Shot noise, quantization		1	
Scene distortion		4	
Optics distortion		3	5
Dissector distortion, drift, jitter		4	
Gimbal readout		7	5
Gimbal alignment		--	7
	RSS	10	10

Attitude Reference Errors

Error Source		Random ($\widehat{\text{sec}}$, 1σ)	Bias ($\widehat{\text{sec}}$, 1σ)
Tracker (star mode)		--	8
Gyros and Software		5	6
	RSS	5	10

Environmental Model

Landmark radius uncertainty (1σ)   500 feet

Cloud cover $\approx$ latitude, longitude dependent probability model

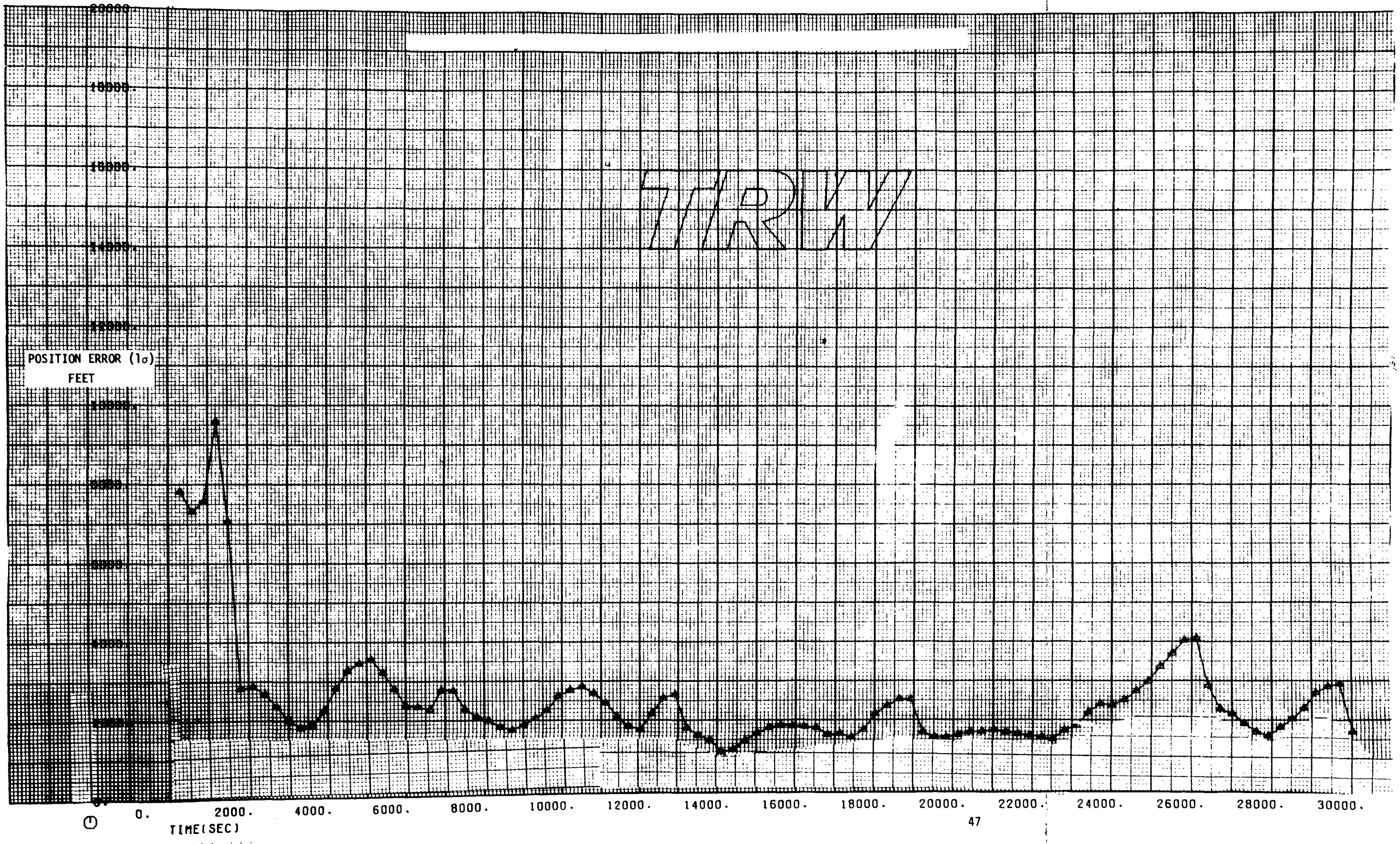


Figure 3-16. Navigation Accuracy for a Combined Landmark Tracker/Horizon Sensor System

TRANSFER TRAJECTORY PERFORMANCE

This section summarizes the results of a study to evaluate the navigation performance along a transfer trajectory that originates in a low circular orbit and proceeds to intercept a tracking satellite at synchronous altitude. This trajectory travels through approximately a 180 degree central angle from injection to intercept in approximately 5.5 hours. Navigation uncertainties along this trajectory were determined for the following navigation systems:

(1) Horizon Sensor System

(2) TDRS (Tracking and Data Relay Satellite) System

(3) Ground Station System (MSFN)

The unknown landmark tracker system and the ground transponder system were not considered because of their range constraints. However, performance of the ground transponder system can be inferred from that obtained for MSFN.

The geometry for the transfer trajectory to intercept a TDRS satellite in geostationary orbit is illustrated in Figure 3-17. The transfer trajectory assumes a single maneuver injection from a 150 nautical mile, 29 degree inclination circular orbit. The post maneuver trajectory is a 250 nautical mile by 19,360 nautical mile orbit with an inclination of 26.8 degrees. This trajectory travels through approximately a 180 degree central angle from injection to intercept in 5.5 hours.

The navigation uncertainties developed during the transfer maneuver were determined assuming the following injection burn characteristics:

Duration ≈ 1600 sec

Acceleration ≈ 5.2 ft/sec^2

Direction ≈ normal to radius vector,
8 degrees out of plane

The IMU error model for the onboard inertial system contained the following sensor error values:

Gyros (1σ):

Bias drift	.01 deg/hr
Mass unbalance	.015 deg/hr/g
Compliance	.005 deg/hr/g^2

Accelerometers (1σ):

Bias	60 μg

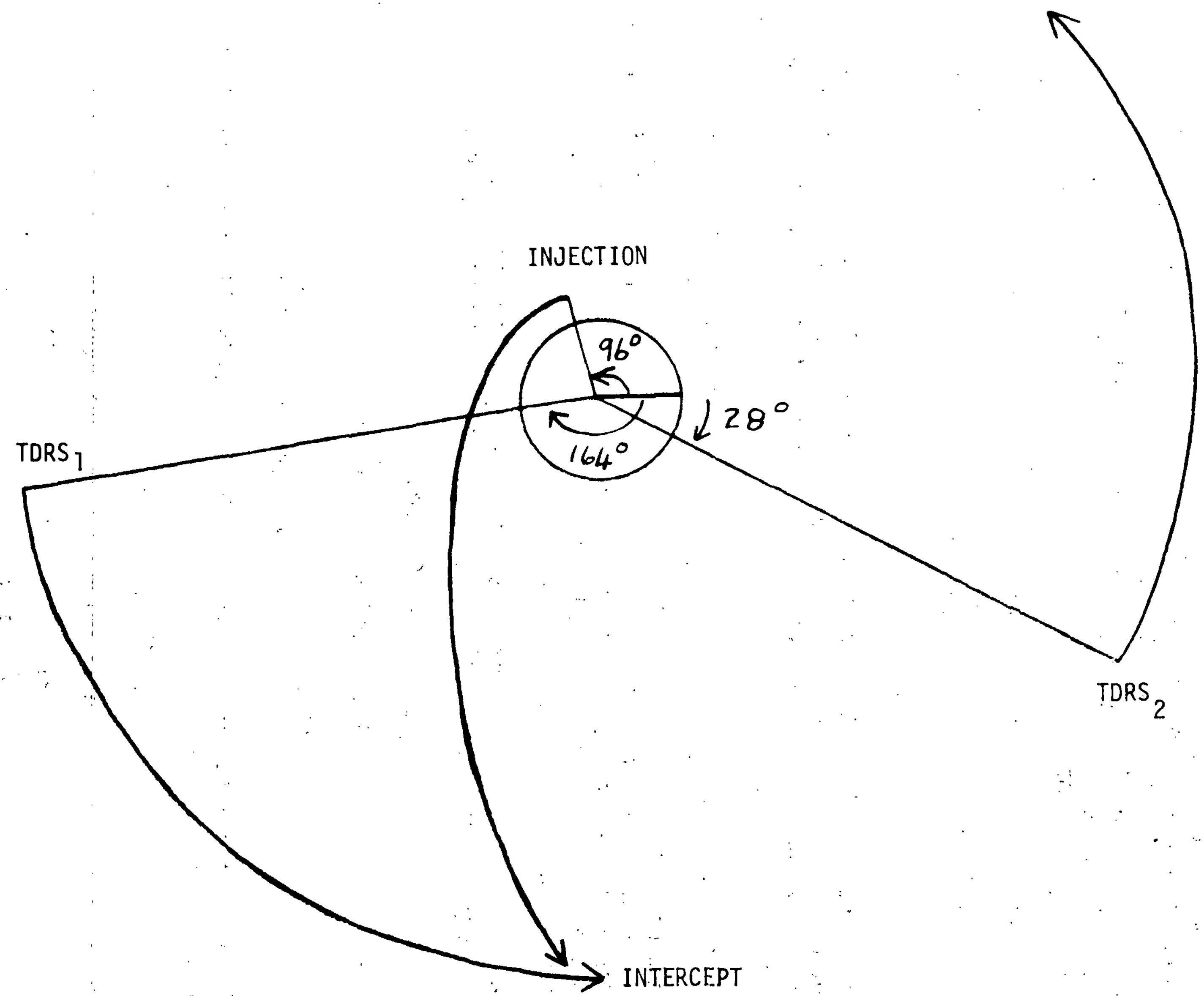


Figure 3-17. Transfer Trajectory Geometry

Scale factor error	34 ppm
Input axis misalinement	40 sec

This IMU error model coupled with an inflight alinement uncertainty of 40 sec (1σ) was applied to the previously defined injection burn profile to generate the covariance matrix for the navigation uncertainties developed during the maneuver. Initial condition uncertainties representative of TDRS tracking were added to the maneuver uncertainties to yield the following covariance matrix.(feet and feet per second) in a U (radial), V (downrange), and W (crossrange) coordinate system:

$$\begin{bmatrix} 1.19{+}7 & -5.35{+}6 & -.63{+}6 & 2.33{+}4 & -5.98{+}3 & -.44{+}3 \\ & 8.38{+}6 & -.30{+}6 & -1.38{+}4 & 6.19{+}3 & -.21{+}3 \\ & & 3.87{+}6 & -.85{+}3 & -.23{+}3 & 3.19{+}3 \\ & & & 4.78{+}1 & -1.07{+}1 & -.60 \\ \text{SYMMETRIC} & & & & 5.13 & -.17 \\ & & & & & 2.67 \end{bmatrix}$$

The injection maneuver covariance matrix was propagated to the transfer trajectory intercept time to determine the navigated state uncertainties assuming no external tracking data. The 1σ state uncertainties at the injection and intercept points are:

	U	V	W	RSS	$\dot{U}$	$\dot{V}$	$\dot{W}$	RSS
Injection	3,449	2,895	1,968	4,914	6.91	2.26	1.63	7.45
Intercept	112,531	162,851	12,172	198,323	16.9	3.8	.3	17.3

Horizon Sensor System

The horizon sensor system considered here consists of three horizon sensors that measure the angle to the earth's horizon with respect to vehicle fixed axes and a stellar inertial reference system that determines the orientation of the vehicle axes in inertial coordinates. The measurements used for navigation are derived by combining the horizon sensor with the onboard estimate of vehicle inertial attitude to form a "star"/horizon angle measurement where the inertial reference is used to form a fictitious star line-of-sight. The geometry of the star horizon measurement is shown in Figure 3-18a. The horizon measurement angle is sensitive to position errors perpendicular to the horizon line-of-sight in the measurement plane defined by the horizon line-of-sight and the vehicle position vector. The individual horizon sensors were oriented at the sighting azimuths given in Figure 3-18b. to provide inplane and crosstrack navigation updates.

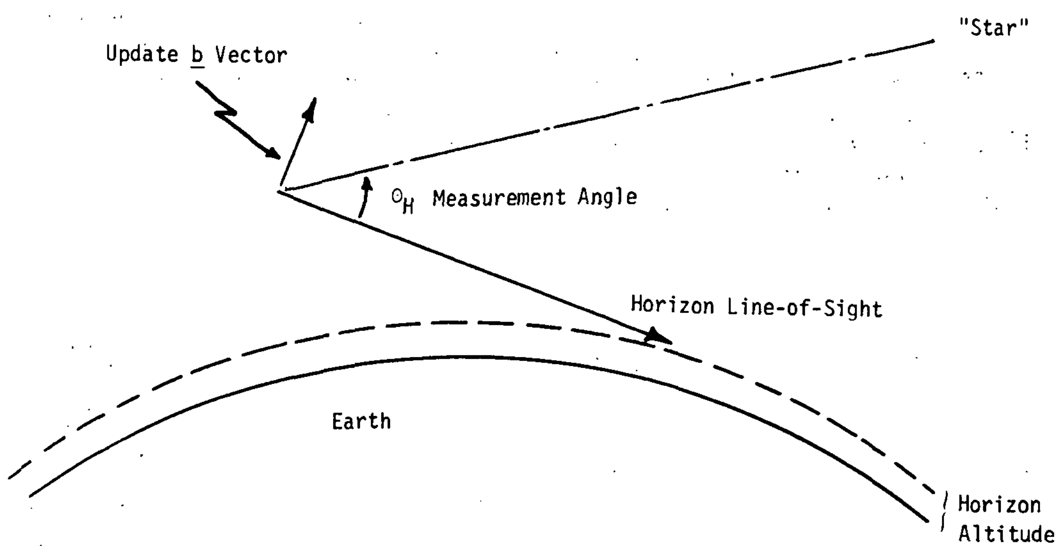


Figure 3-18a. Horizon Angle Measurement Geometry

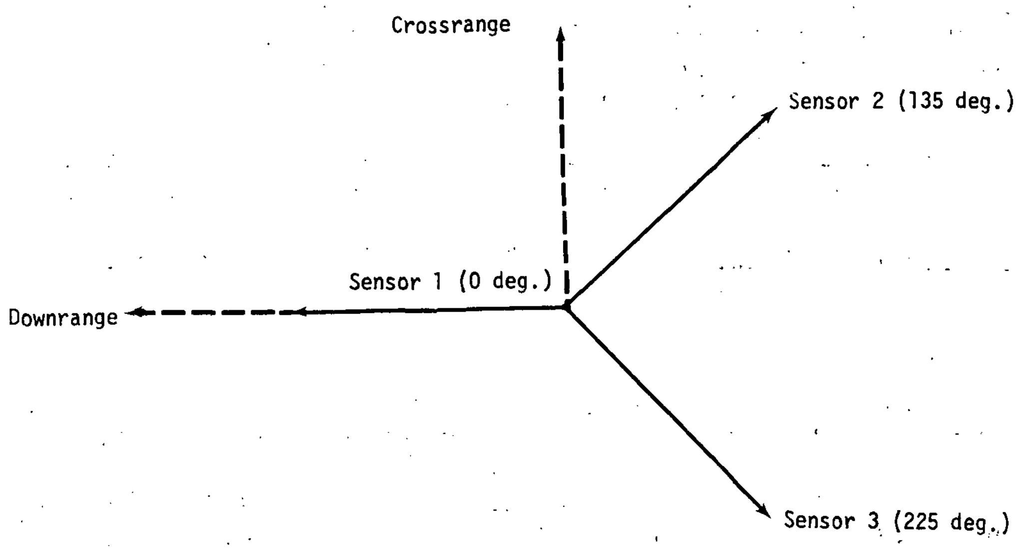


Figure 3-18b. Horizon Sensor Sighting Azimuths

The horizon sensor system error model used in the navigation analysis is presented in Table 3-9. The real world model represents the true sensor system and environmental errors modeled in the program. The filter model defines the measurement error model for the assumed onboard navigation filter.

The dominant error in the horizon sensor measurement is the uncertainty in the horizon altitude. Onboard mathematical models can be used to compensate for the known variations as a function of latitude, season, spacecraft altitude, etc. The residual altitude error after compensation is modeled in the program as a fixed bias although a more representative model is a random process with both a time and spatial correlation. This deficiency in the program environment error model was compensated for to some extent by assuming an independent altitude bias for each sensor.

The data rate used in the navigation error analysis consisted of one measurement per minute from each of the three horizon sensors. Tracking intervals consisted of the first and third 15 minute segment of each hour (total of 30 minutes per hour) during the entire transfer trajectory.

The total (1σ) position and velocity navigation uncertainties are presented in Figures 3-19a and 3-19b. The curves labeled "Propagation" show the time history of the injection maneuver uncertainties propagated to intercept without navigation updates. The horizon sensor tracking schedule is defined along the time axis of each figure.

Navigation uncertainties for updates from the horizon sensor system are plotted for two values of the horizon altitude bias sigma (σ_H) - 3,000 ft and 1,500 ft - to illustrate the sensitivity to this error source. The lower solid curves represent the navigation uncertainty at that time point on the transfer trajectory. The upper dashed curves represent the navigation uncertainty propagated to intercept assuming no additional measurement data.

The following comments pertain to the significant results of the performance analysis:

(1) The position error at intercept was approximately 58,000 ft (RSS) for the case with the altitude uncertainty σ_H equal to 3,000 ft. The position error reduces to 41,000 ft (RSS) when σ_H is reduced to 1,500 ft in the second case. This navigation performance falls somewhat short of the desired 30,000 ft (RSS) position accuracy.

(2) The horizon sensor navigation updates provided essentially no improvement in the crossrange navigation errors at intercept as shown by the crossrange uncertainties in Table 1. The growth of the inplane errors was significantly contained.

(3) The error sources that most significantly contributed to the navigation uncertainties were the horizon altitude bias and the inplane horizon sensor bias. The navigation

Table 3-9. Horizon Sensor System Error Model

PARAMETER	REAL WORLD	FILTER
Initial State Error Covariance	Injection Maneuver Covariance	Diagonal Covariance U 3,000 ft 7.0 fps V 3,000 ft 2.2 fps W 2,000 ft 1.6 fps
Gravity Uncertainty (1σ)		
• Earth (μ_e)	2 ppm	Not Modeled
• Moon (μ_m)	2 ppm	Not Modeled
Attitude Reference Error (1σ)		
• Bias - per axis	10 $\widehat{sec}$	Not Modeled
Horizon Altitude Bias (1σ) (per sensor)	1500 ft	Not Modeled
Horizon Sensor Error (1σ)		
• Bias	10 $\widehat{sec}$	Not Modeled
• Noise	50 $\widehat{sec}$	84 $\widehat{sec}$

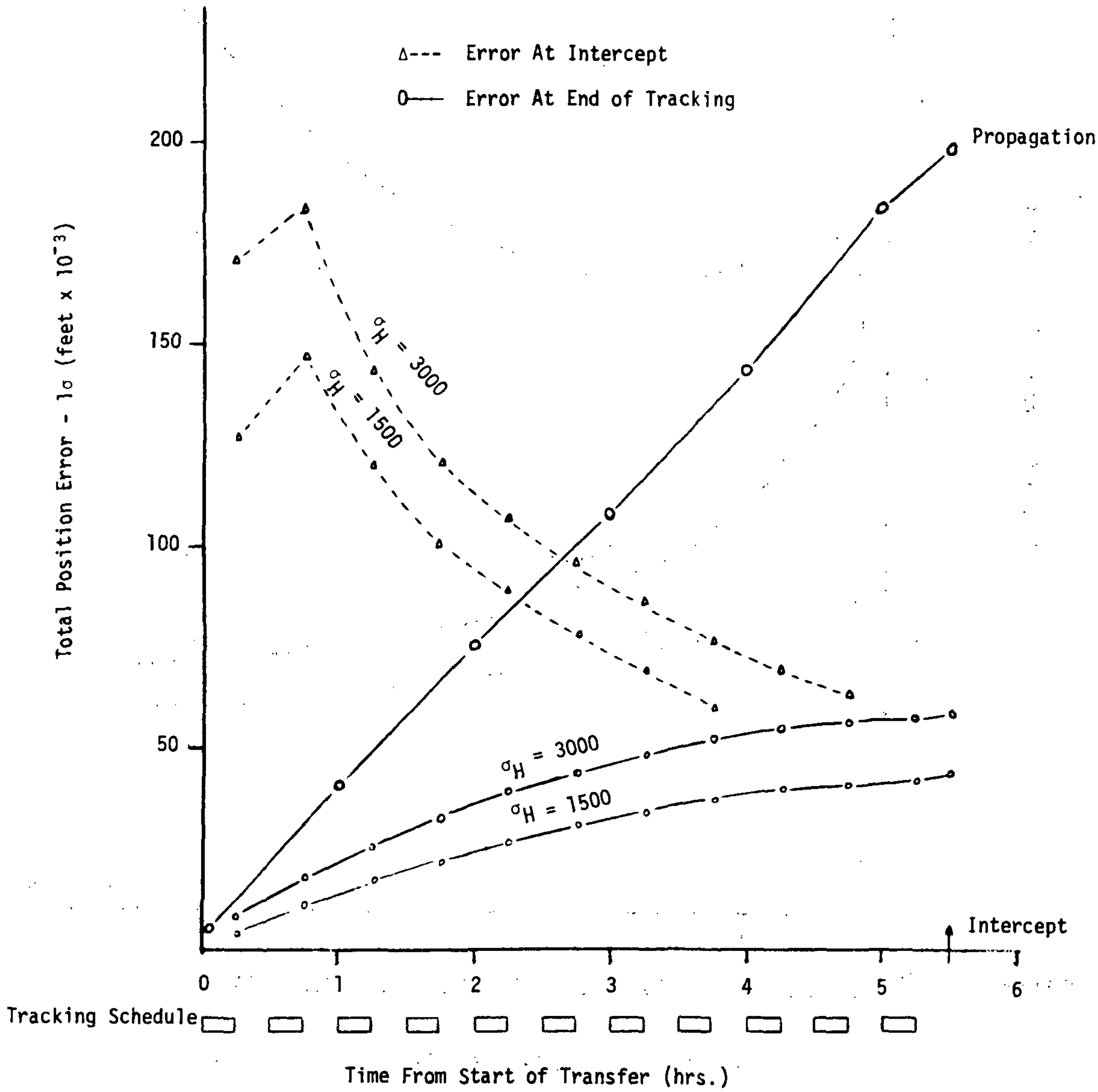


Figure 3-19a. Horizon Sensor Navigation Performance (Position)

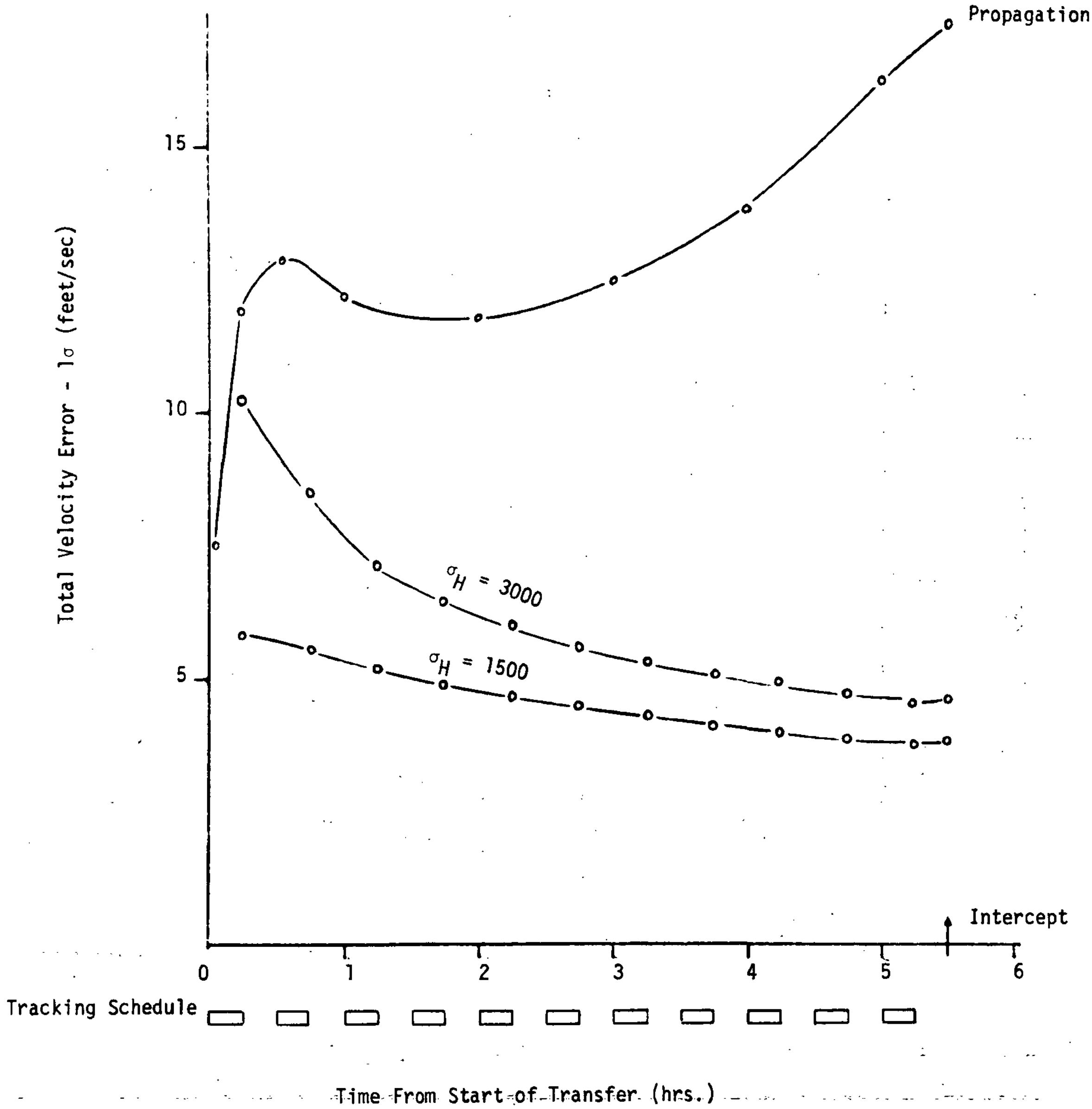


Figure 3-19b. Horizon Sensor Navigation Performance (Velocity)

performance might be improved by including an estimate of a measurement bias in the filter configuration, although the one-half orbit trajectory arc may not be sufficient to make the biases observable. It is felt that studies to evaluate this bias estimation should be performed on a simulation where a sophisticated horizon altitude error model can be mechanized.

TDRS (Tracking and Data Relay Satellite System)

The TDRS navigation system is composed of a configuration of one to three satellites in synchronous orbit with inclinations less than 5 degrees. These satellites act as relays from the ground stations to the user spacecraft. As such, the basic measurements are (1) the total range from the ground station, and (2) the sum of the range rate between the ground and the TDRS and the range rate between the TDRS and the user spacecraft. Because the TDRS is essentially stationary with respect to the ground stations, the range rate measurement is essentially the range rate between the TDRS and the user spacecraft. The field-of-view of the TDRS is approximately a 13 degree half cone angle around the vector to the center of the earth. This restricted field-of-view significantly limits the interval of tracking coverage during the transfer trajectory as discussed later in the navigation performance section. Table 3-10 contains the TDRS error model used for this trajectory.

The relative geometry of the transfer trajectory and two TDRS satellites is shown in Figure 3-17, and the TDRS coverage is shown in Figure 3-20. $TDRS_2$ could provide only a small amount of tracking coverage during the first part of the transfer. $TDRS_1$ can track the vehicle for approximately the first hour of the transfer at which time coverage is lost for the remainder of the coast to intercept. The analysis results presented here represent range and range rate tracking from only $TDRS_1$ and incorporated only during the first and last 15 minute interval for the first hour of the transfer trajectory. The data rate during these two tracking periods was one measurement pair (range and range rate) per minute. Additional data from $TDRS_1$ or $TDRS_2$ did not improve the performance; however, this may have been the result of not properly weighting the measurement data.

The position and velocity 1σ uncertainties for tracking from the one TDRS are presented in Figures 3-21a and 3-21b. The significant results of the performance analysis are:

(1) After the first 15 minutes of tracking, the total position error propagated to intercept is approximately 46,000 ft (RSS). The second interval of tracking at the end of the hour reduces the error at intercept to 26,000 ft (RSS).

(2) The navigation error components show that the TDRS provides essentially no improvement to the crossrange uncertainty, as might be expected from the tracking geometry.

(3) The dominant error source is the TDRS state vector uncertainty.

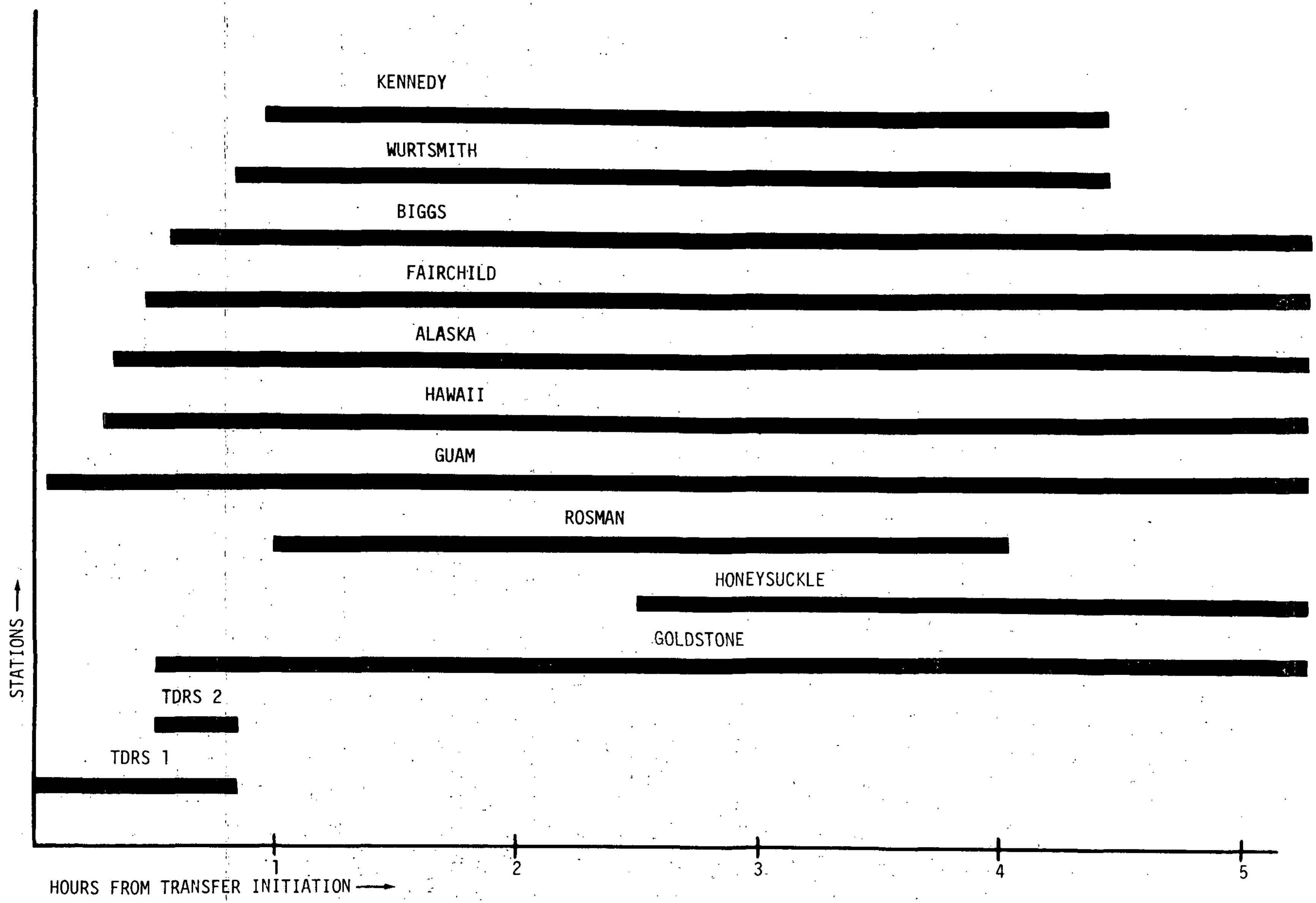


Figure 3-20. Transfer Orbit Coverage

Table 3-10. TDRS System Error Model

PARAMETER	REAL WORLD	FILTER
Initial State Error Covariance	Injection Maneuver Covariance (Section 3.)	Diagonal Covariance U 3,000 ft. 7.0 fps V 3,000 ft. 2.2 fps W 2,000 ft. 1.6 fps
Gravity Uncertainty (1σ)		
● Earth (μ_e)	2 ppm	Not Modeled
● Moon (μ_m)	2 ppm	Not Modeled
Range Error (1σ)		
● Bias	90 ft.	Not Modeled
● Noise	140 ft.	980 ft.
Range Rate Error (1σ)		
● Bias	.002 fps	Not Modeled
● Noise	.056 fps	.392 fps
TDRS State Error	Covariance Matrix in UVW (ft., fps)	Not Modeled

$$\begin{bmatrix} .2414+4 & -.1252+6 & .300+7 & .9114+0 & .4936-2 & -.1494+0 \\ & .8204+6 & -.9950+4 & -.5952+2 & -.7228+0 & -.2561+0 \\ & & .1760+6 & .9559+0 & -.8971+0 & -.1384+0 \\ & & & .4324-2 & .5249-4 & .8753-4 \\ & & & & .1250-4 & .1389-4 \\ & & & & & .9637-3 \end{bmatrix}$$

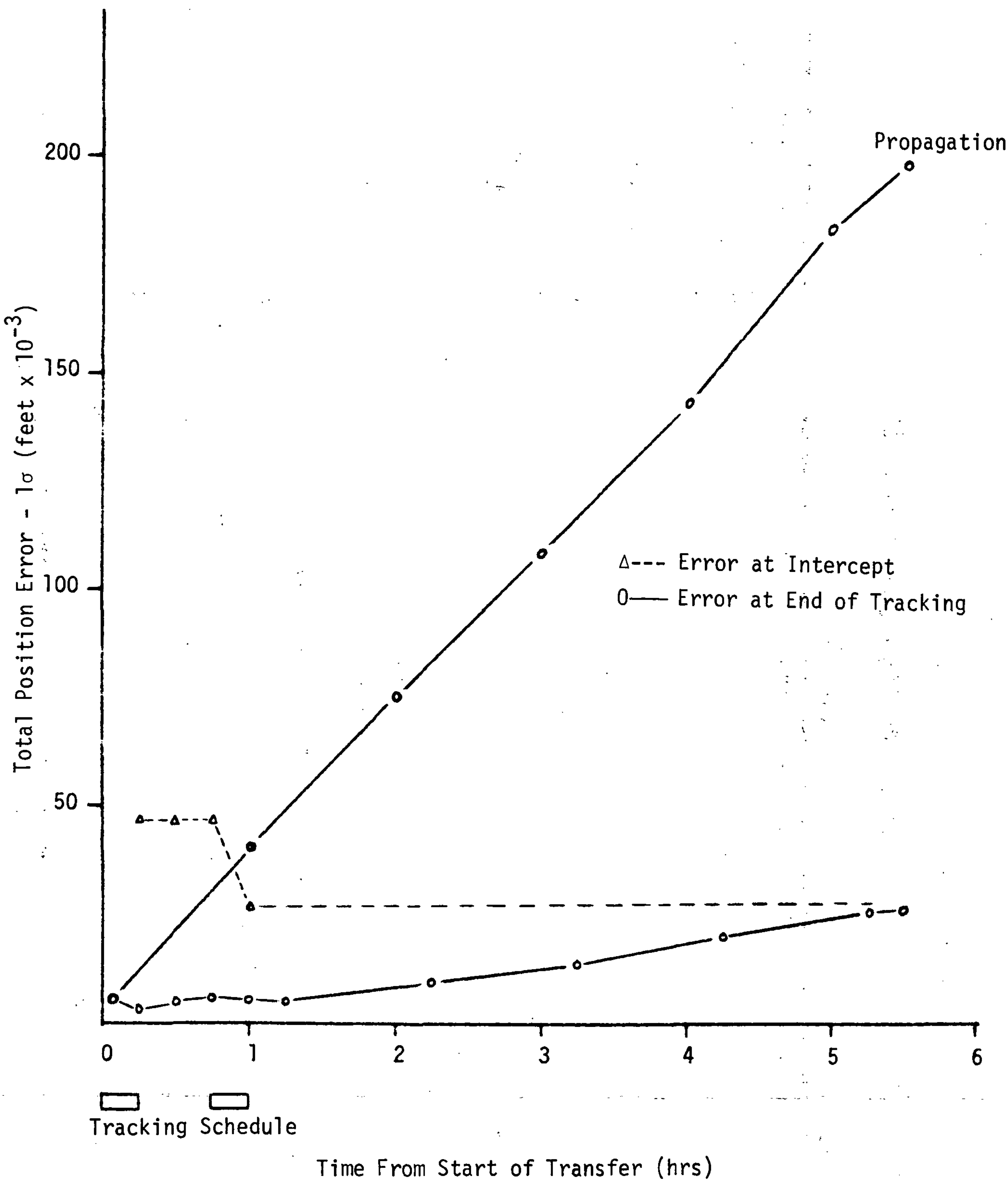


Figure 3-21a. TDRS System Navigation Performance (Position)

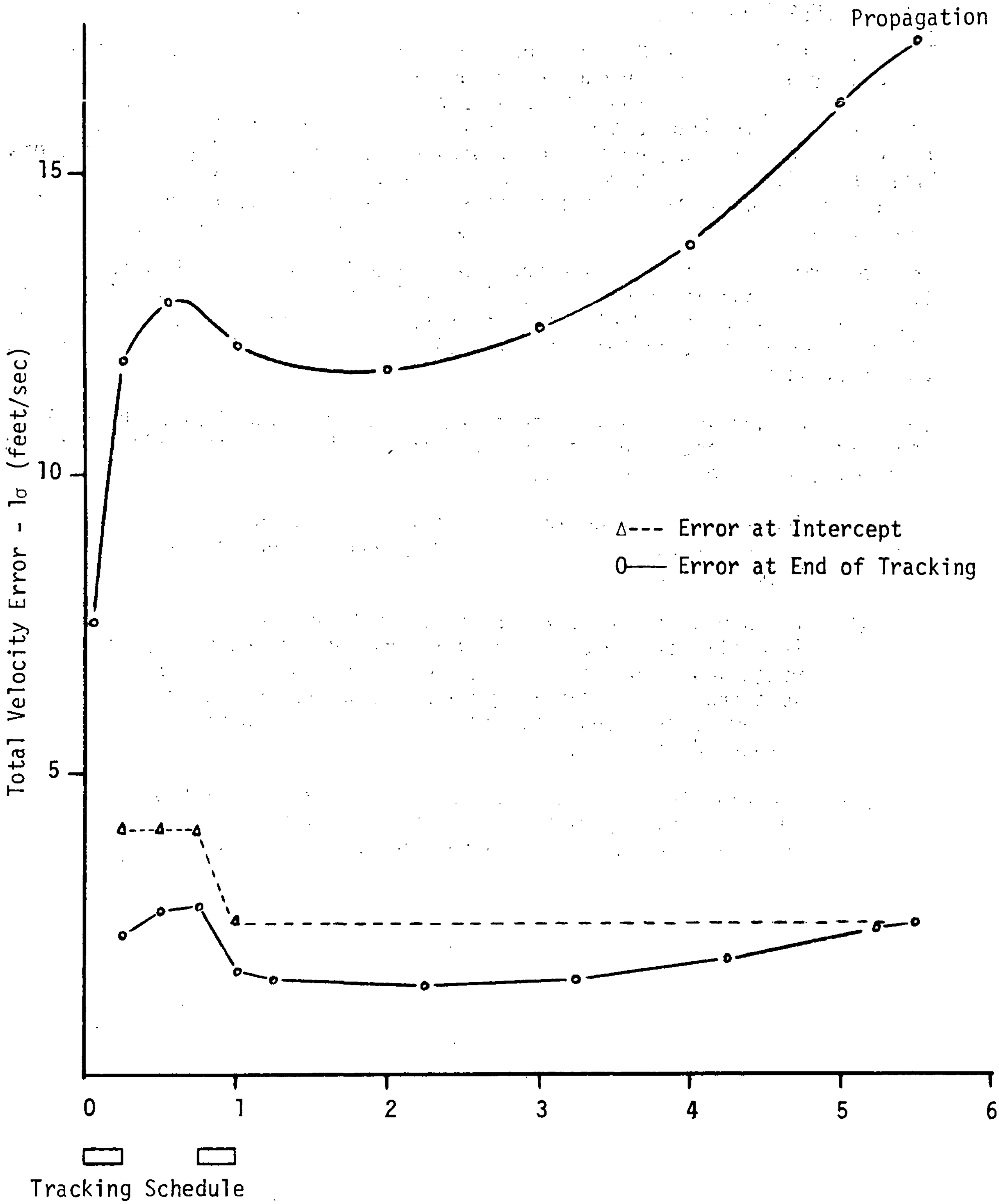


Figure 3-21b. TDRS System Navigation Performance (Velocity)

MSFN (Manned Spaceflight Network)

The utilization of the MSFN network for navigation during the transfer trajectory assumes that range and range rate data from the unified S-band (USB) tracking systems would be processed on the ground and the resultant state vector uplinked to the spacecraft. The nature of the transfer trajectory allows long periods of tracking from individual ground stations. This is indicated by Figure 3-20 which includes MSFN stations and the 7 station locations considered earlier for the ground transponder system. Table 3-11 contains the MSFN error model used for this trajectory.

The performance analysis evaluated the navigation accuracy obtained when tracking from one and two MSFN stations - Goldstone and Honeysuckle. The data rate used consisted of one measurement pair (range and range rate) per minute from the station for the intervals defined by the tracking schedule at the bottom of Figure 3-22a.

The position and velocity 1σ uncertainties for tracking from both the MSFN stations are presented in Figures 3-22a and 3-22b. The significant results of the performance analysis are:

(1) The total position error at intercept was approximately 2,200 ft (RSS) for the tracking schedule defined in Figure 3-22a.

(2) The navigation performance presented for the MSFN system is applicable to a system of ground transponders supplying range and range rate data to be processed onboard the spacecraft. The limiting factor in a ground station system is tracking coverage for a system with state-of-the-art data accuracies.

(3) Tracking from Goldstone alone provides an accuracy at intercept of approximately 15,700 ft, with the largest component in the crossrange direction (12,700 ft).

Table 3-11. Ground Station System (MSFN) Error Model

PARAMETER	REAL WORLD	FILTER
Initial State Error Covariance	Injection Maneuver Covariance	Diagonal Covariance U 20,000 ft 20 fps V 20,000 ft 20 fps W 20,000 ft 20 fps
Gravity Uncertainty (1σ)		
• Earth (μ_e)	2 ppm	Not Modeled
• Moon (μ_m)	2 ppm	Not Modeled
Range Error (1σ)		
• Bias	60 ft	Not Modeled
• Noise	30 ft	210 ft
Range Rate Error (1σ)		
• Bias	0	Not Modeled
• Noise	.002 fps	.014 fps
Station Location Error (1σ)		
• Altitude	140 ft	
• Latitude	1 $\widehat{sec}$	Not Modeled
• Longitude	.8 $\widehat{sec}$	

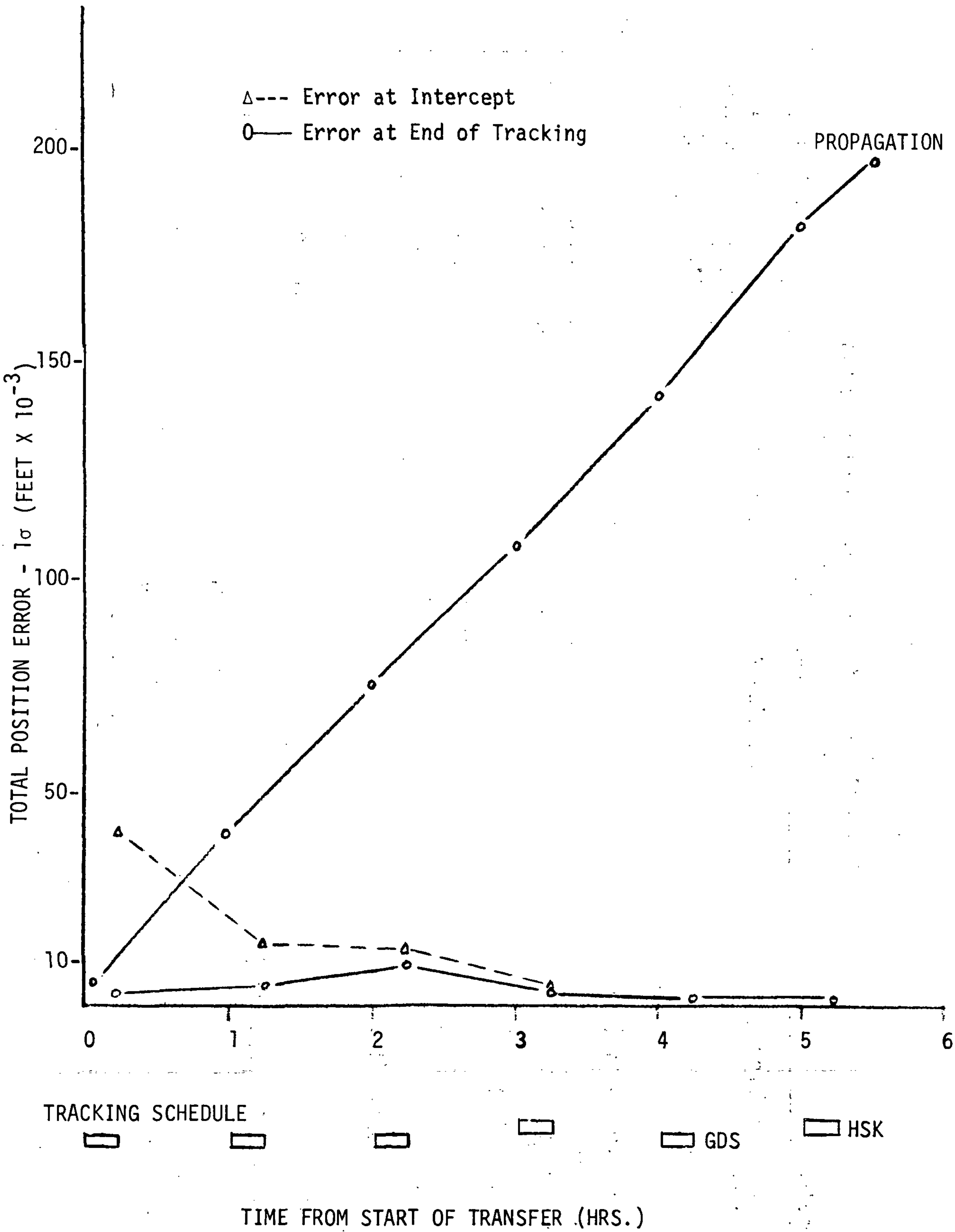


Figure 3-22a. Ground Station System Navigation Performance (Position)

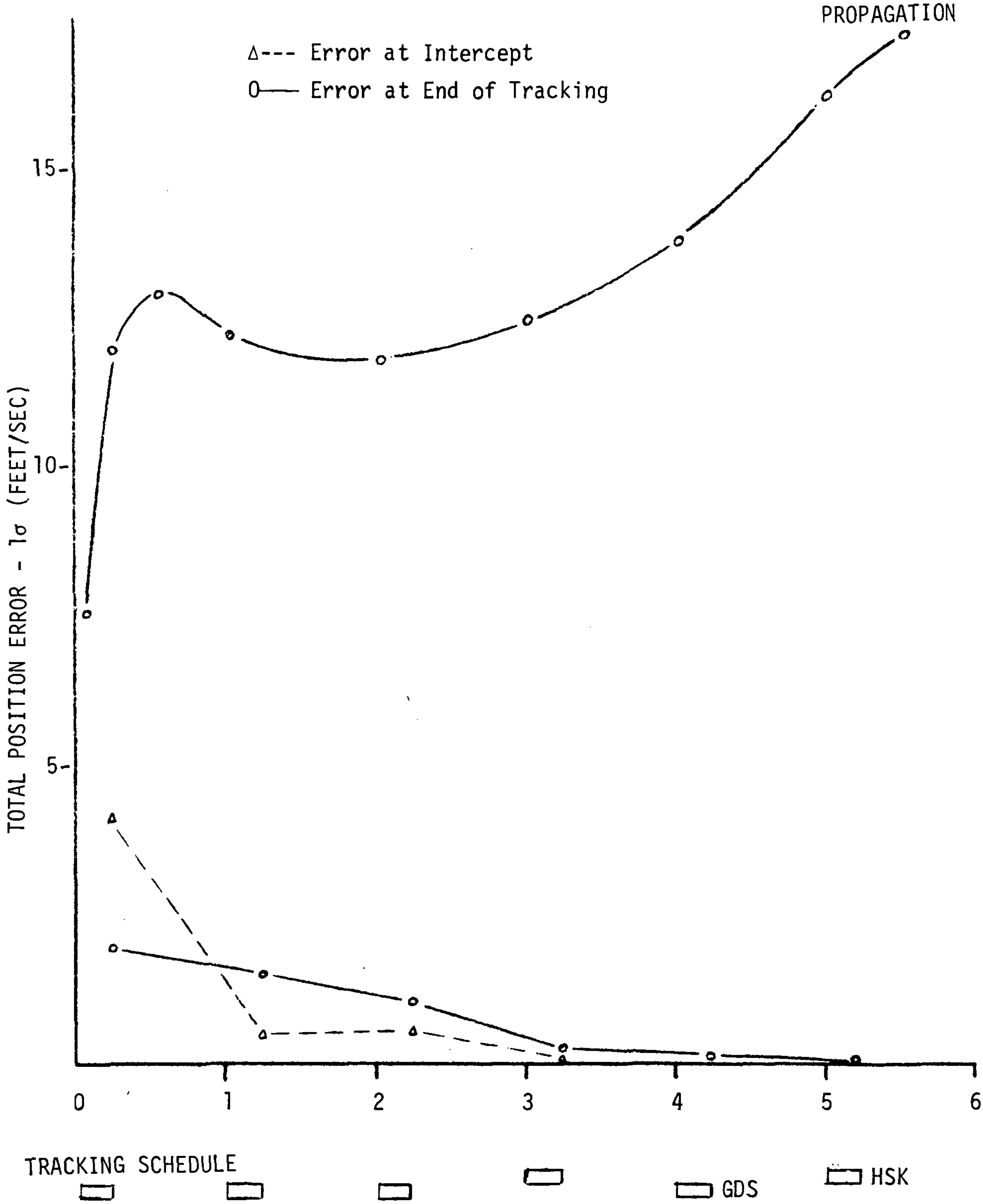


Figure 3-22b. Ground Station System Navigation Performance (Velocity)

MULTIFUNCTION REQUIREMENTS AND CAPABILITIES

The requirements and guidelines which were developed for the space station are assumed to be applicable to the two earth orbital missions considered in this study. These data have been obtained from requirements published by MSC and the space station phase B contractors McDonnell Douglas (MDC) and North American Rockwell (NAR). The applicable documents are References 31 through 46.

In general, the orbit navigation function is to be a near autonomous operation that meets performance requirements for the least cost and complexity. However, high complexity in the form of hardware is less a problem if the hardware can be used to provide or support other functions (without sacrifices in the performance, reliability, etc., of those other functions).

MSC PERTINENT SPACE STATION GUIDELINES AND CONSTRAINTS

- A Tracking and Data Relay Satellite System will be operational.
- Two-way communications between ground and any subsatellite to be provided by station.
- System and mission status will not be necessarily transmitted to ground on real-time basis.
- Ground support equipment is to be minimized.
- Attitude restrictions to maintain communications should be minimized.
- Continuous voice communications with ground not required.
- Station will have onboard tracking capability.
- Free Flying Modules (FFM's) are under ground control when not within station tracking range.
- Ground tracking capability will be provided during early phases (at least until onboard navigation capability is verified).
- The primary flight modes are: local vertical during normal operations and inertial for experiment support and docking.

PERFORMANCE REQUIREMENTS

Orbit Navigation (1σ accuracies)

	NAR	MDC
Attitude:	± 1500 ft	± 790 ft
In-Track:	± 3800 ft	± 1640 ft
Cross-Track:	± 2200 ft	± 1100 ft
Velocity:	3.5 fps	0.4 fps

The above shall be provided during Routine Operations in support of experiments (attached and integral), orbit makeup, and Shuttle or Tug updates (pre-undocking).

Navigation shall be accurate enough to allow inertial pointing of communications equipment to within ± 5° (MDC).

Orbit makeup requirements are based on Jacchia 240 nm altitude model atmosphere and maneuvers are to be concurrent with CMG desaturation. Nominal time between makeup maneuvers not to be less than 12 hrs (MDC).

RENDEZVOUS NAVIGATION

MSC Requirements:

Coverage area:		100 ft - 1100 nm
Range errors:	Noise	30 ft
	Bias	30 ft
Range rate errors:	Noise	.13 fps
	Bias	.03 fps
Angle (LOS) measurement:		.2° (0-300 nm)
Relative velocity range:		± 2350 fps
Acquisition time:		5 sec
Data Rate:	Range	6/min
	Range Rate	60/min

NAR Requirements:

Coverage Area	Range	Range Rate	Angle
<100 ft	0.5 ft (1σ)	--	--
100 ft - 1000 ft	1.0 ft (1σ)	--	--
1000 ft - 20 nm	500 ft	0.5%	--
20 nm - 450 nm	3000 ft	0.5%	--

MDC Requirements: (3σ)

Coverage Area	Range	Range Rate	Angle
0 - 110 nm	0.1%	TBD	1.4°
110 nm - 1000 nm	0.1%	--	--

SURVEILLANCE AND TRAFFIC CONTROL

Tracking (all cooperative spacecraft except RMS):

	Range	Range Rate	Angle
NAR:			
<100 ft	0.5 ft (1σ)	--	--
100 ft - 1000 ft	1.0 ft (1σ)	--	--
1000 ft - 450 nm	500 ft	0.5 fps	--
MDC:			
0 - 110 nm	0.1%	TBD	1.4°
110 nm - 1000 nm	0.1%	--	--

The area of coverage for targets between 20 nm and 450 nm is a radial distance in the orbital plane and for targets between 1000 ft and 20 nm the coverage is spherical (NAR).

Tracking of multiple targets not required and detached modules shall be constrained to a range of 450 nm (NAR).

Deployment, retrieval, stationkeeping (inertial position):

NAR: 1.0 nm (spherical)

Tracking/transponding shall be available as follows (NAR):

Station-shuttle to 1100 nm
Station-FFM to 450 nm

Station must be able to track and transpond simultaneously (NAR).

RMS (REMOTE MANEUVERING SUBSATELLITE) EXPERIMENT SUPPORT

Plasma Wake Experiment

This experiment consists of one RMS and two inflated balloons positioned within 400 - 1000 ft in front of the Space Station. One balloon will probably be spherical (60' dia.) and the other cylindrical (120' long x 60' dia.). Translation and plane changes will be required of the RMS. The station must know the position and velocity of the balloons and the relative distances and closure rates between them and the RMS. In addition, the RMS must be tracked with the following accuracy when within 300 ft of the Space Station:

	Range	Angle
MSC:	1.0 ft	--
MDC:	1.0 ft	0.2°

The frequency of experiments is to be one earth orbit per day.

The RMS will be remotely controlled by the station and will have its own stabilization and propulsion subsystems. During experiments it will be required to move in and out of the balloon wake area.

Radio Occultation Experiment

Several RMS', possibly 4-6, will be required for this experiment. The Space Station must track and provide rendezvous support for these subsatellites over a relative range of 10 ft to the horizon.

ATTITUDE REFERENCE DETERMINATION

MDC navigation performance requirements (3σ) to meet attitude reference requirements:

Inclination uncertainty	± .02°
Nodal longitude uncertainty	± .02°
Orbit regression rate	± 10%
Orbit rate uncertainty	± .003%
Orbit angle	± .02°

Attitude Reference Requirements (MDC):

	Fine Control (1σ)		Coarse Control (1σ)	
Orientation	Att.	Att. Rate	Att.	Att. Rate
Horizontal (X-POP/LV)	± .02°	± .001°/s	± .2°	± .001°/s
Inertial (exp. support)	± .01°	± .001°/s	--	--

COMMUNICATION REQUIREMENTS

To & from ground via TDRS:

MDC:	K-Band	voice, video, data
NAR:	VHF	voice, data
	K-Band	voice, video, ranging, data

To & from ground direct:

MDC:	S-Band	data, voice, video, ranging
NAR:	S-Band	voice, video, data, ranging

To & from FFM's:

MDC:	K-Band	data, ranging, video
NAR:	S-Band	data, ranging, voice, video

To & from shuttle:

MDC:	VHF	data, ranging, voice
NAR:	S-Band	data, ranging, voice

NAR communications data characteristics for the Space Station are shown in Table 4-1. MDC requirements are as follows:

To & from ground via TDRS:

K-Band:	120 Kbps - PM subcarrier 10 Mbps - FM subcarrier

To & from ground direct:

S-Band	120 Kbps - PM subcarrier 10 Mbps - FM subcarrier

To & From FFM's:

K-Band	10 Kbps - PM subcarrier 1 Mbps - PCM/PSK

To & from Shuttle:

VHF	10 Kbps - PCM/PSK

SUMMARY OF REQUIREMENTS AND GUIDELINES

The data presented in the previous paragraphs has been summarized to aid in the development and evaluation of the candidate mechanizations for orbit navigation.

- Communications - Requirements are those of a high data rate user necessitating a K-band or S-band system to handle voice, ranging, and data.
- Attitude Reference - Requirement based upon attitude control is an accuracy of .01 degree (1σ); more stringent requirements could result from the navigation system.
- Rendezvous Navigation - Requirement for coverage out to 1100 nautical miles (MSC,MDC) is excessive based upon Apollo experience and Skylab plans. The NAR coverage requirement of 450 nautical miles appears to be reasonable. The data requirements for navigation include line of sight angular data as well as range or range and range rate.
- Stationkeeping Navigation - Spherical coverage requirement from 1000 feet to 20 nautical miles. The coverage requirement for ranges less than 1000 feet is assumed to be constrained to the longer range capability.
- Functional Redundancy - Although this capability is not explicitly called for, it is desirable.

SYSTEMS CAPABILITIES

The potential capabilities of the systems to perform the functional requirements as defined in the previous paragraphs have been assessed based upon the navigation performance evaluations and the mechanization analyses.

MSFN and TDRS - As an integral part of the communication system, these systems will perform this function. Systems do not provide any additional functional support.

Ground Transponder - System can support rendezvous with range and range rate measurements but must use a gimballed antenna or an optical device to provide angle measurements required for rendezvous. Some functional redundancy is available during rendezvous from the comparison of the range, range rate, and angle measurements.

Horizon Sensors - Can provide attitude reference redundancy in two axes in a local vertical coordinate system; with a state vector and one gyro, the complete inertial attitude can be established. Internal redundancy is provided by a four sensor head configuration.

Landmark Tracker - Proposed mechanization would use same optics for attitude reference and navigation. System can support rendezvous with angle measurements on a flashing beacon, but needs a range measurement in addition to complete the required measurement set.

Table 4-1. Communications Data Characteristics (NAR)

		RF Channel	Voice	Television	System TLM	Computer Data	Experiment Data	Text/Graphics	Command Data	EVA TLM	Facsimile	Ranging: Measure	Ranging: Respond
From Space Station To	Detached RAM	S-band	(1) 300-4000 Hz						10 kbps			0.5 mbps	
	Shuttle orbiter	S-band	(1) 300-4000 Hz		50 kbps							0.5 mbps	0.5 mbps
	MSFN ground terminal direct	S-band	(3) 300-4000 Hz	4.5 MHz	500 kbps	500 kbps	2.0 mbps	1.0 kbps		200 bps			0.5 mbps
	Ground terminal via TDRS	VHF	(1) 300-4000 Hz		10 kbps								
	Ground terminal via TDRS	K-band	(3) 300-4000 Hz	4.5 MHz	500 kbps	500 kbps	2.0 mbps	1.0 kbps		200 bps	0.5 MHz		0.5 mbps
	EVA	VHF	(1) 300-4000 Hz										
To Space Station From	Detached RAM	S-band	(1) 300-4000 Hz	2.9 MHz	50 kbps	500 kbps	Part of system TLM						0.5 mbps
	Shuttle orbiter	S-band	(1) 300-4000 Hz						1.0 kbps			0.5 mbps	0.5 mbps
	MSFN Ground Terminal Direct	S-band	(4)* 300-4000 Hz			500 kbps		1.0 kbps	1.0 kbps			0.5 mbps	
	Ground Terminal via TDRS	VHF	(1) 300-4000 Hz						1.0 kbps				
	Ground Terminal via TDRS	K-band	(4)* 300-4000 Hz			500 kbps		1.0 kbps	1.0 kbps				
	EVA	VHF	(1) 300-4000 Hz							200 bps			

*One of the four voice channels - ground to MSS - is a high-fidelity channel 30-10,000 Hz for entertainment.

SYSTEM MECHANIZATION

A basic mechanization scheme for each of the five navigation systems is developed through the consideration of alternate mechanizations and redundant configurations for each subsystem element. The types of alternate designs considered includes gimballed or body fixed, sensor selection, and redundancy level. The mechanization choice is then determined from the total cost, performance and multifunctional capability of the system.

The total functional requirements for the tug vehicle and the space station include communications and attitude reference. These functions must be provided regardless of which navigation system is employed. However, there is a direct interaction of these two requirements and some of the navigations systems considered. The TDRS and MSFN navigation systems are closely tied to the communication system. To perform the orbit navigation function with these two systems, a range and range rate transponder must be integrated with the communication system. The horizon sensor system and the unknown landmark system require an attitude reference system. Although the attitude reference system is considered as part of the total functional requirements rather than an integral part of the navigation system, some discussion of a mechanization is warranted.

ATTITUDE REFERENCE

The requirements placed on the attitude reference system are obtained from the considerations of mission length, attitude control system requirements, and navigation system requirements. The ten year duration of the space station indicates the need for a long life time system which can be readily maintained or replaced. The potential candidates for the system include a strapdown IMU with component redundancy or redundant gimballed IMUs with a star tracker or star mapper for star sightings to bound the error growth.

The strapdown system with redundant components offers advantages in reliability and data processing for both rate measurement and failure detection and isolation. Failure detection can be performed with single redundancy (four gyros) and failure isolation can be performed with double redundancy (five gyros). A strapdown IMU with six gyros has been considered by MIT (Reference 17) and a four gyro system has been considered by TRW (Reference 18). A redundant gimballed system can provide the same failure detection and isolation as a strapdown system with five gyros. However, after a single gyro failure, the gimballed system is no longer redundant, while the strapdown system retains its failure detection capability with the four remaining gyros. Although the acquisition cost will be greater than that for a gimbal system, the total cost considering reliability and maintenance appears to be less for the strapdown system.

The accuracy requirements for the attitude reference system depend upon the requirements for pointing control (given in the Multifunction Analysis as 0.01 degrees) and the navigation system requirements. The performance analysis used an accuracy of 0.003 degrees for the unknown landmark tracker navigation system. Thus this system may require additional techniques and sensors to obtain the required accuracy. Again, from the performance analysis of the tug transfer trajectory to synchronous orbit, the horizon sensor system also requires an accuracy greater than that of the space station pointing control.

TDRS, TRANSPONDER, AND MSFN SYSTEMS

The TDRS, transponder, and MSFN prime candidate earth orbital navigation systems employ communication techniques* and thereby are similar in many general respects. These common considerations will be discussed first, followed by block diagram analyses of each system.

Coverage

As shown in Figure 5-1, each of the candidate configurations relies on one or more communication links between the orbiting spacecraft (Space Station or Tug) and the ground. The TDRS system requires a direct line-of-sight (LOS) between the spacecraft and the relay satellite and between the satellite and the ground. The transponder and MSFN systems require a direct LOS between the spacecraft and the ground.

Referring to Figure 5-2, the coverage provided by a given transponder or MSFN station depends upon 1) the pattern of the ground antenna (including pointing maneuverability if narrowbeam), 2) local horizon obstructions, and 3) spacecraft altitude. Given terrain cross-sections at the ground site (as in Figure 5-2a), a plot of the coverage contours for various-altitude orbits can be obtained (as in Figure 5-2b). If the ground track of the spacecraft falls within the coverage contour, a line-of-sight exists, and tracking measurements can be performed for updating navigation parameters. The coverage time of a particular orbital pass over the ground site depends upon how the spacecraft ground track intersects the contour. For a given set of coverage contours, increased probability of coverage can be obtained by placing the spacecraft in a higher orbit, provided the link margins are

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* Skin Tracking techniques such as radar are not candidates because of their inflexible configurations (primarily ground-fixed) and relatively low precision.

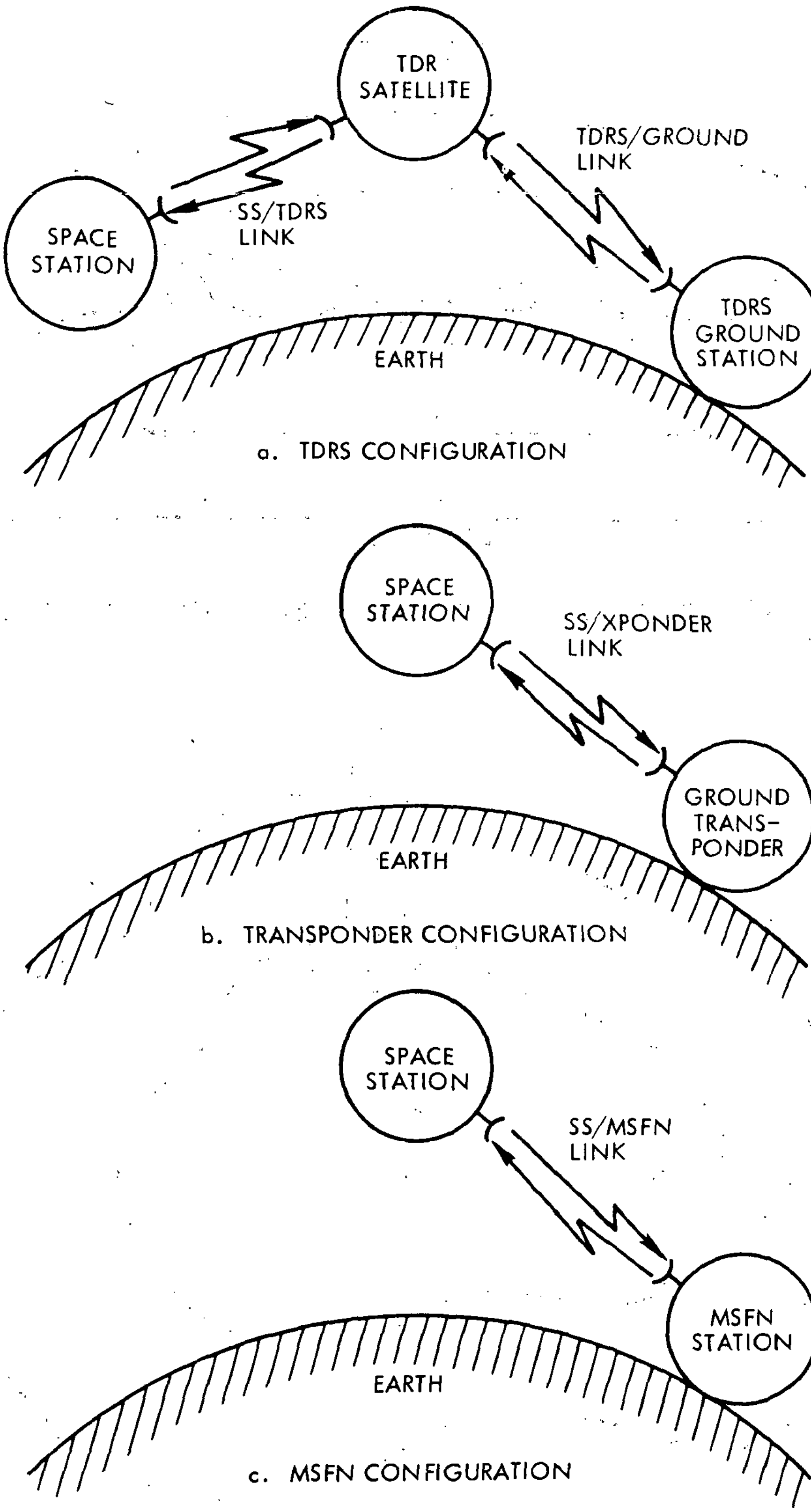
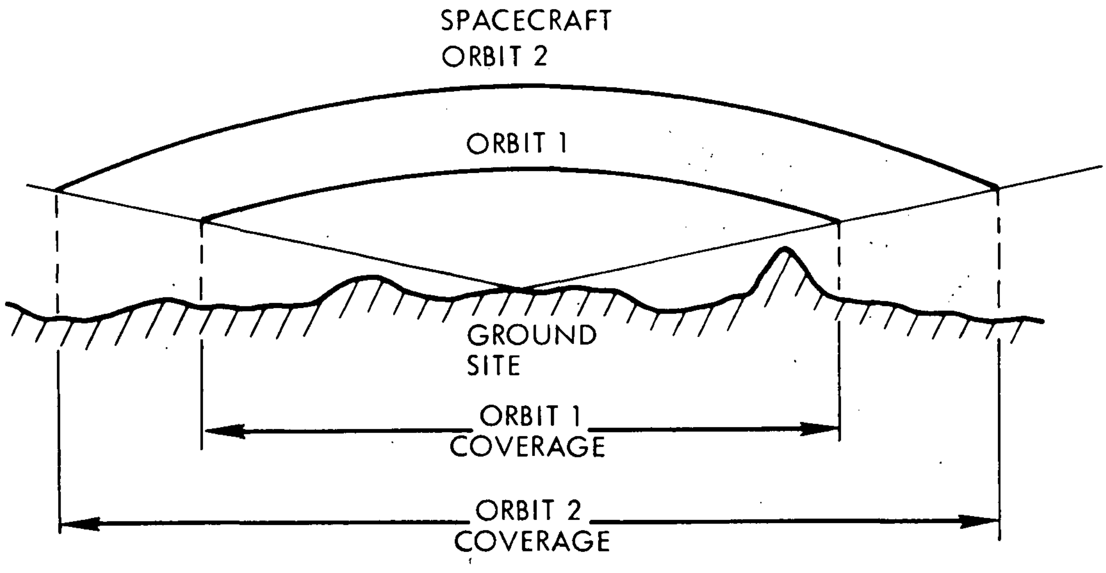
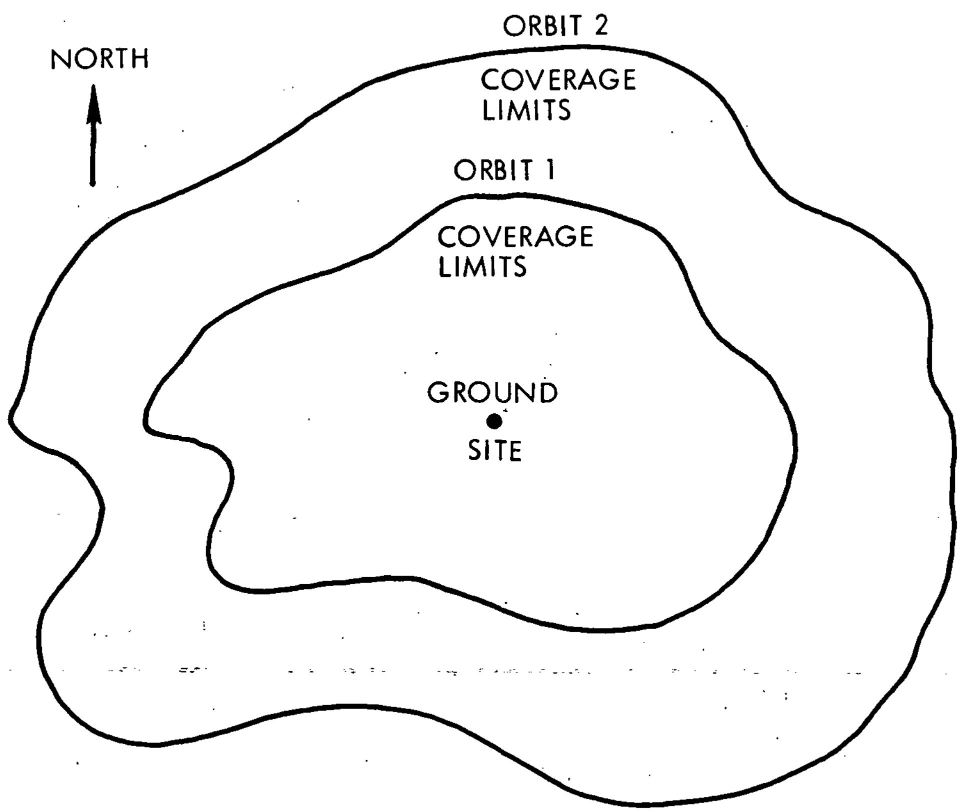


Figure 5-1. Candidate Communication/Tracking Links



a. CROSS SECTION OF TYPICAL GROUND STATION COVERAGE



b. TYPICAL GROUND STATION COVERAGE CONTOURS

Figure 5-2. Typical Ground Station Coverage adequate for the higher orbit. From an alternate viewpoint, fewer ground sites would be required to provide a given overall level of coverage as the orbit altitude is increased.

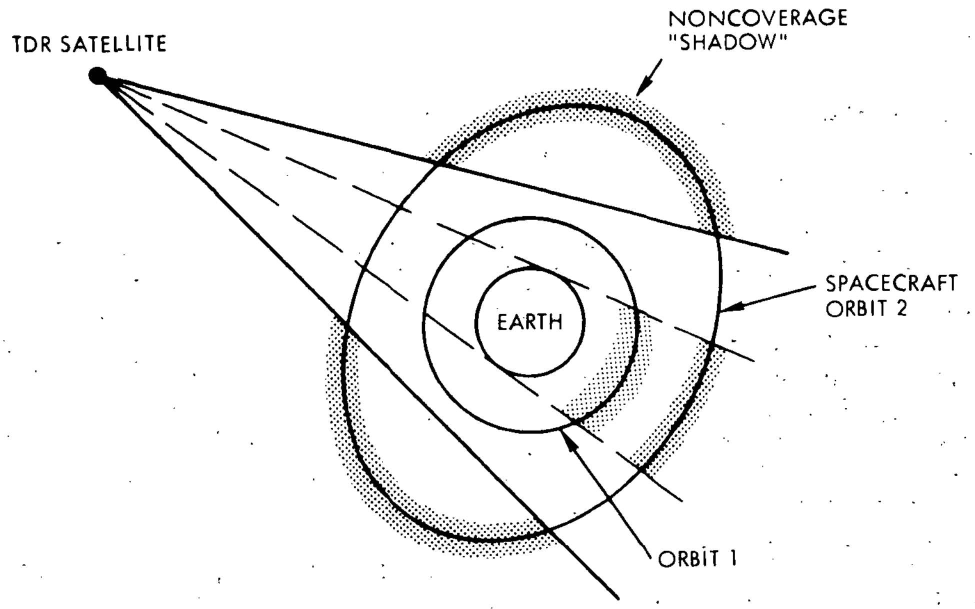
As shown in Figure 5-3, the TDRS system is composed of one or more relay satellites in geosynchronous orbits; each satellite is always within line-of-site of one or more ground stations. The coverage of a user spacecraft provided by a given relay satellite depends upon 1) the satellite-to-space craft antenna pattern (including pointing maneuverability) and 2) spacecraft altitude. In Figure 5-3a, the relay satellite coverage pattern is symmetrical about the earth and is intended to provide line-of-sight coverage of user spacecraft in low earth orbit up to some maximum altitude. Even with a single relay satellite, the available coverage for low altitude users is significantly greater (assuming adequate link margins) than with the transponder or MSFN configurations; noncoverage occurs only when the user is "shadowed" by the earth. By suitably locating one or more additional satellites and adding ground stations where necessary, low-altitude users (e.g., Space Station) can be provided with continuous coverage. High-altitude users (e.g., Space Tug) would not be so fortunate, since significant portions of their orbits could lie beyond the relatively narrow beamwidth of the TDRS antennas. Such users might be served better by the transponder or MSFN systems.

The preceding discussion has implied that a continuous availability of communications is desireable. This is particularly the case with manned missions, such as Apollo, which require significant ground-based support. But, depending on mission objectives and operational procedures, the Space Station or Tug tracking mechanizations may require significantly less coverage. For example, if on the order of only two to five minutes are required for a navigation fix, if only about one fix per orbit is required, and if a fix is not required on every orbit, the relative advantages or disadvantages of the TDRS, transponder, or MSFN configurations becomes much less distinct. The transponder or MSFN configurations may be able to provide adequate coverage with a relatively small number of ground sites. Or the TDRS configuration may be able to provide adequate coverage for spacecraft in orbits that would otherwise be considered to be too high.

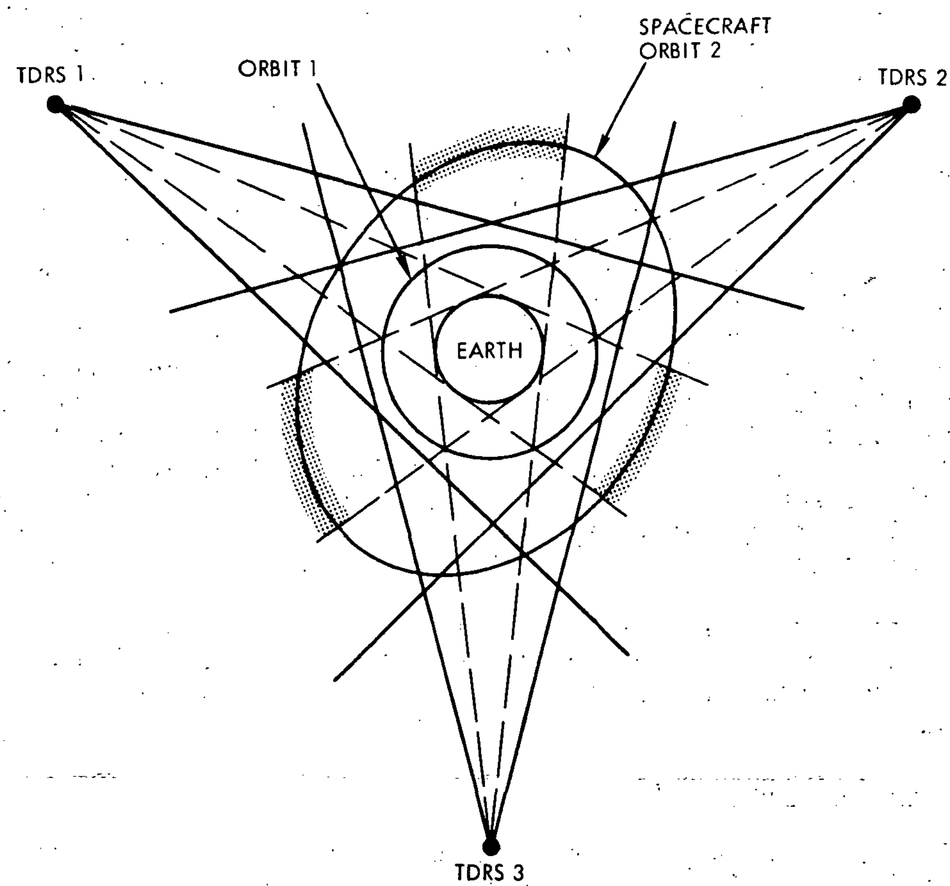
On the other hand, the longer the navigation system has to operate without an update, the larger will be the error in its calculated position between fixes. Thus, some compromise is required in establishing the required coverage.

Range and Range Rate Systems

Any of several CW ranging system mechanizations can be considered as candidates for each of the TDRS, transponder, or MSFN systems. As noted in the preceding paragraphs, these radio ranging techniques rely on communication links between the orbiting spacecraft and the ground or other spacecraft. Navigation parameters are determined primarily by measuring the



a. ONE SATELLITE COVERAGE



b. THREE SATELLITE COVERAGE

Figure 5-3. Typical TDRS Coverage

range (or distance) and range rate (or velocity) along the line-of-sight (LOS) between the spacecraft and the ground (or another spacecraft); in a few cases, steerable antenna gimbal angles are also used, where the antenna is pointed at and tracks the "target."

As with radar, the range measurement is determined by the time required for a recognizable signal pattern to propagate between the ends of the communication link. In a radar system, a signal "pulse" is transmitted to a target, where it is either merely reflected (passive skin tracking) or amplified and retransmitted (active transponder). The time elapsed from when the pulse is sent until the "echo" is received is directly proportional to the (two-way) range to the target. The CW systems are essentially equivalent to radar systems, except that continuous rather than pulse-type signals are usually employed, signal reflection is not used, and some form of time and/or signal synchronization is required between the transmitting and receiving ends of the link. Such systems can be one-way or two-way, coherent (signal phase preserved) or noncoherent (signal phase is not maintained).

There are two commonly used types of range rate measurement. In a phase coherent system, range rate is usually derived from the doppler shift (rate of change of phase) of the received signal. In a noncoherent system, doppler shift is not meaningful, so range rate must be obtained from some other measurement, such as by averaging the differences between successive range measurements. But such a derived range rate measurement would not be independent of the range measurement.

A simple CW ranging system is diagrammed in Figure 5-4a. This is a one-way, noncoherent system which relies upon carefully calibrated reference signals at the transmitter and receiver. At some initial time, the references are aligned, or a precisely-known difference is determined. If the two references have sufficiently small drift and random bias errors, and if the delay in the transmitter/receiver combination is similarly well known and stable, any measurable deviation from exact alignment of the received and reference signals at the receiving station is a measure of the range between the two stations.

The one-way system in Figure 5-4a would probably not be coherent, since the two references would have to have highly accurate open loop tracking of each other to allow adequate doppler measurements. Such an open loop system would require exceptionally stable references.

Figure 5-4b shows a two-way, noncoherent ranging system which differs from a radar system primarily by operating on two frequencies*: the master

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* Note that some systems (e.g., the Apollo Rendezvous Radar) are not truly radars in the sense employed here but rather are CW ranging systems.

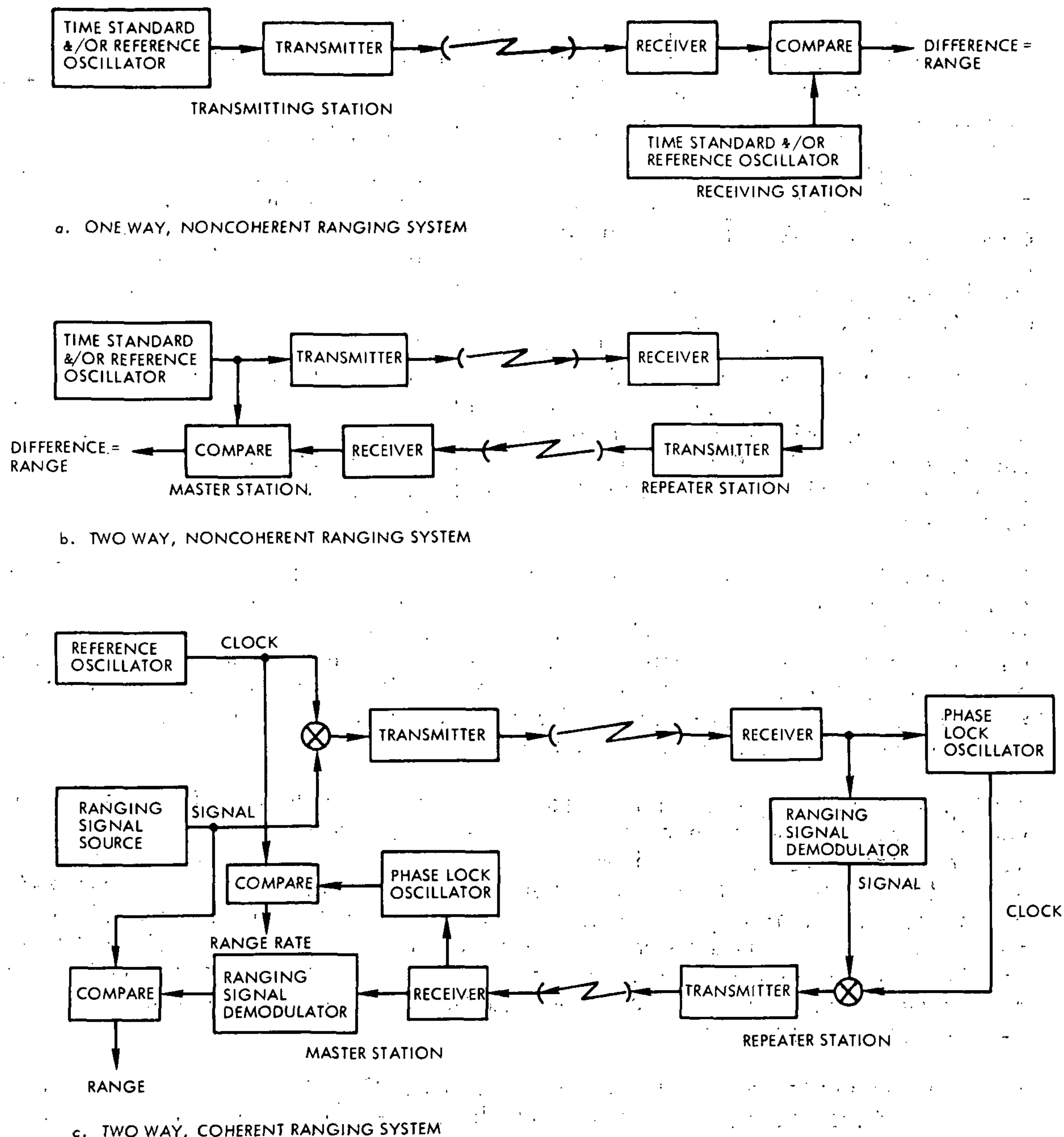

Figure 5-4. CW Ranging Systems

station transmits to the repeater station on one frequency channel, while the repeater replies on another channel, merely "turning around" what it receives. This system, being a form of transponder, has the advantage that all of the sophisticated range- and range-rate-determining equipment is located at the master station. The repeater can be very simple in mechanization. As with the one-way system, no attempt is made to preserve signal phase, so independent range rate is not available.

The two-way, coherent ranging system outlined in Figure 5-4c is more complex than either of the two noncoherent systems considered. This complexity allows simultaneous, independent determination of range and range rate by using a composite ranging signal; the signal to be used for range determinations is combined with a clock. At the repeater station, the clock signal is recovered coherently in a phase lock circuit; the ranging signal is recovered separately. The reconstructed signals are recombined and retransmitted back to the master station, where the received signals are similarly reconstructed. The range and range rate measurements are derived by comparing the transmitted with the received signals. While such a system is relatively complex, it offers the significant advantages of being able to operate with essentially independent range and range rate accuracy requirements.

Range and Range Rate Techniques

There are two primary CW ranging techniques in use today and projected for use during the Space Station era: tone and digital code. The predominant tone techniques are 1) fixed frequency tones and 2) swept (chirp) tone. Systems employing a fixed tone or tones are more common and more readily understood than any other type of ranging system. Swept tone systems have an advantage with regard to acquisition time but require more complex techniques for resolving range and range rate ambiguities. The accuracies achievable by these techniques are identical, the fundamental difference being acquisition time, but the accuracy of a fixed tone system is easier to maintain. Current and future applications of tone techniques include

1. Cubic CR-100 series ground transponder systems
2. Goddard Range and Range Rate (GRARR) tracking and telemetry system
3. Apollo VHF ranging and voice systems
4. Apollo Rendezvous Radar.

Two popular digital ranging techniques employ 1) PRN (pseudo-random noise) codes and 2) BINOR (binary optimum ranging) codes. The BINOR code sequence is equivalent to a system using fixed tones having a frequency ratio of two between adjacent tones. The BINOR code has the advantage of easier generation and faster acquisition than the PRN code. Current and future applications of digital techniques include

1. PRN Code

    a. Apollo USB and SGLS communication and tracking systems

    b. Motorola AROD ground transponder system

    c. JPL/DSN tracking and telemetry systems

2. BINOR Code

    a. TRW USCANS ranging, voice, and telemetry system

A comprehensive listing of candidate tracking systems for use on the Space Shuttle is given in Reference 1.

A system employing fixed tones can achieve the same range and range rate accuracies as a digital code system. A digital code can be used where a precise synchronizing signal is required for data decoding, or where security is a requirement. Where these are not required for a given system, it is not necessary and may not be desirable to employ a digital coding technique, with its attendant increase in equipment complexity.

Both tone and digital ranging systems usually have in common the feature of obtaining the fine range measurement by measuring the phase of a sinusoid or square-wave signal and comparing with a reference signal. The range resolution increases as the frequency or bandwidth of the signal increases.

The PRN codes are especially well adapted to applications where a precise synchronization signal is required for data decoding or where security is required. Their unique correlation properties also make them well suited for combatting the effects of short- or long-path multipath and interfering signals. A system employing fixed tones is capable of the same range and range rate accuracies as a system employing a digital code. Tone systems can be made less susceptible to multipath by using FM modulation and by increasing the individual tone modulation indices to large values; if the signal strength is higher than the interference, an FM receiver treats other signals as cochannel interference and, with a suitably high capture ratio, suppresses unwanted signals.

Generalized Mechanization

Any of the several candidate ranging mechanizations can be derived from the configuration diagrammed in Figure 5-5. Note that only two-way, transponding systems are being considered; significant system parameters are identified (in parentheses) in the figure. In the interrogator or master station, the ranging signal generator supplies the ranging signals for transmission to the transponder or slave (turn-around) station. These signals may or may not be coherent with the transmit carrier frequency. In the transponder, the received ranging signal is filtered (e.g., in a tone or code tracker) to improve the signal-to-noise ratio (S/N) and is remodulated for transmission back to the interrogator. In a noncoherent turn-around system, the transponder carrier oscillator is free-running (e.g., a crystal oscillator), while in a coherent system, the carrier oscillator is phase locked to the received carrier. The returned signal is processed in the interrogator in similar fashion. The reconstructed ranging signal is compared with a reference to obtain the range measurement; if circuit delays are known ahead of time, the reference can be delayed by an appropriate amount to simplify the interpretation of the range indication, or suitable delay compensation can be made by the computer operating on the range data. Range rate information is derived by comparing the doppler shift in the received carrier with a coherent reference; the reference can be multiplied by the transponder turn-around ratio to simplify a direct comparison.

Fixed Tone Mechanization

The candidate two-way, coherent, fixed-tone mechanization is shown in Figures 5-6 and 5-7. Here, the ranging signal consists of a group of coherent tones, each of which modulates an RF carrier. The range ambiguity resolution is equal to the period of the lowest frequency tone, while the range measurement resolution and accuracy is determined by the period of the highest frequency tone. The intermediate frequency tones are used to resolve the range ambituities of higher tones.

The optimum frequency ratio between adjacent tones is about two, but such a tone ratio maximizes the number of tones and thereby requires more complex receiver and range extraction circuits. The simplest receiver requires the fewest number of tones and hence a maximized tone ratio; this ratio should not exceed eight for reliable ambiguity resolution in the presence of noisy signals.

PN Code Mechanization

The candidate two-way, coherent, PN code mechanization is shown in Figures 5-7 and 5-8. Here, the ranging signal is a repetitive binary sequence having pseudo-noise (PN) properties. The range ambiguity is determined by measuring to the nearest bit the phase of the received sequence. The fine range or accuracy measurement is obtained from the phase of the bit clock associated with the received sequence.

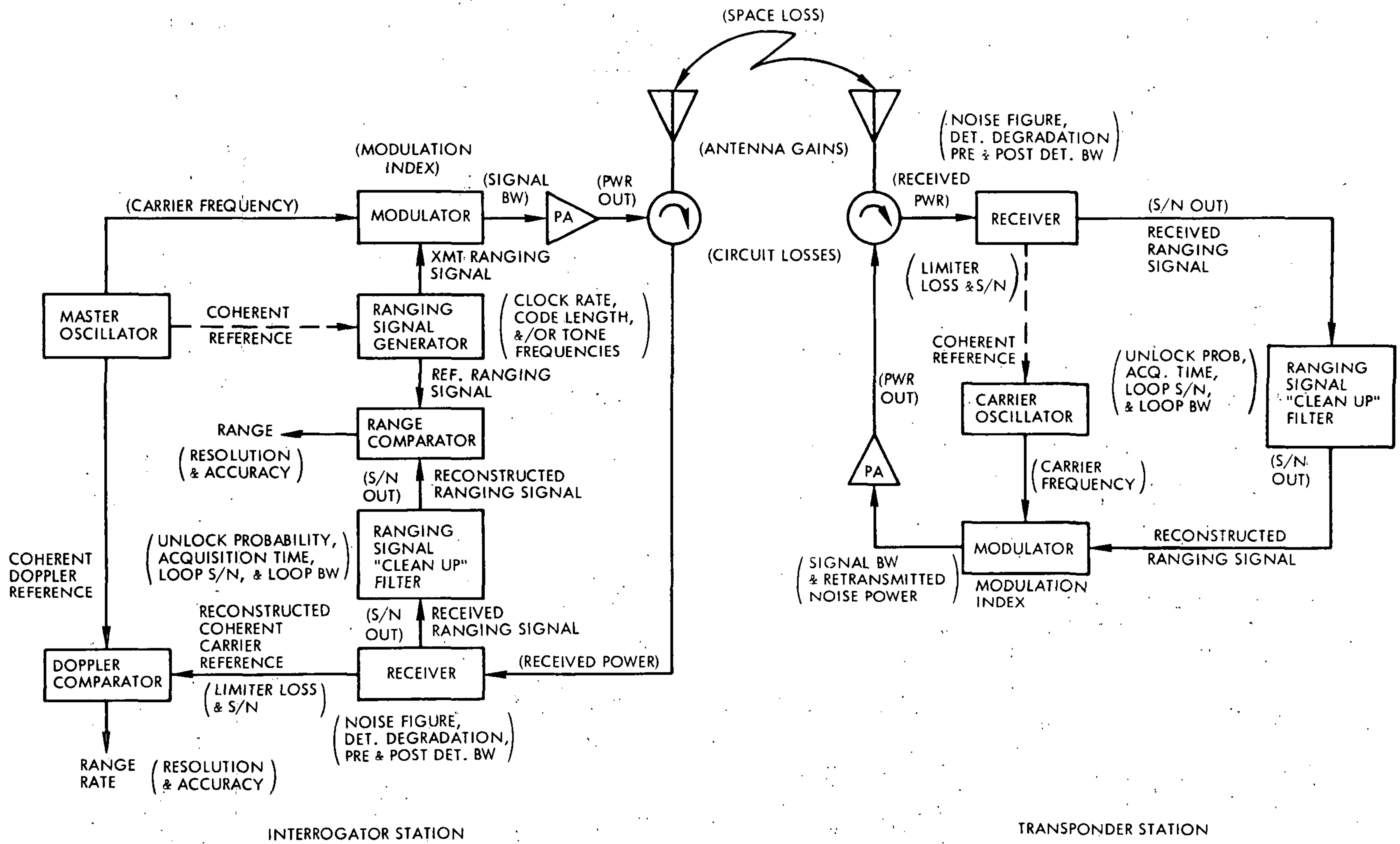

Figure 5-5. Basic Ranging System Baseline Block Diagram (Showing Significant Parameters)

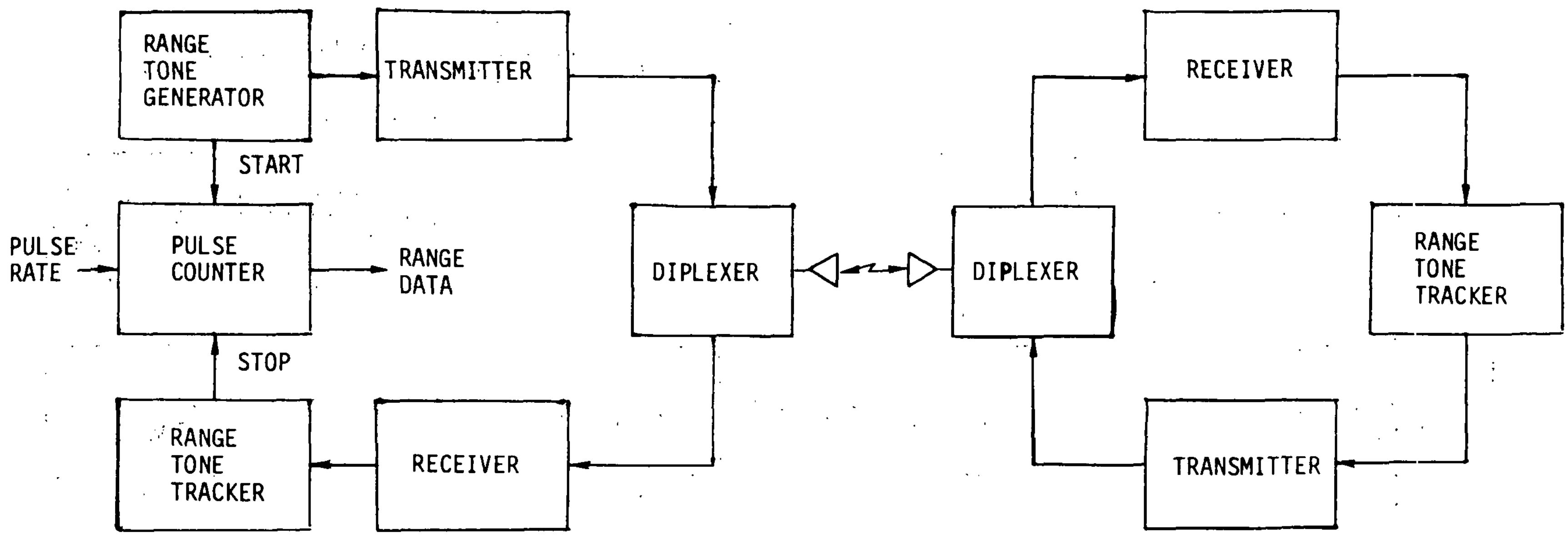

Figure 5-6. Fixed-Tone Range Measurement Mechanization

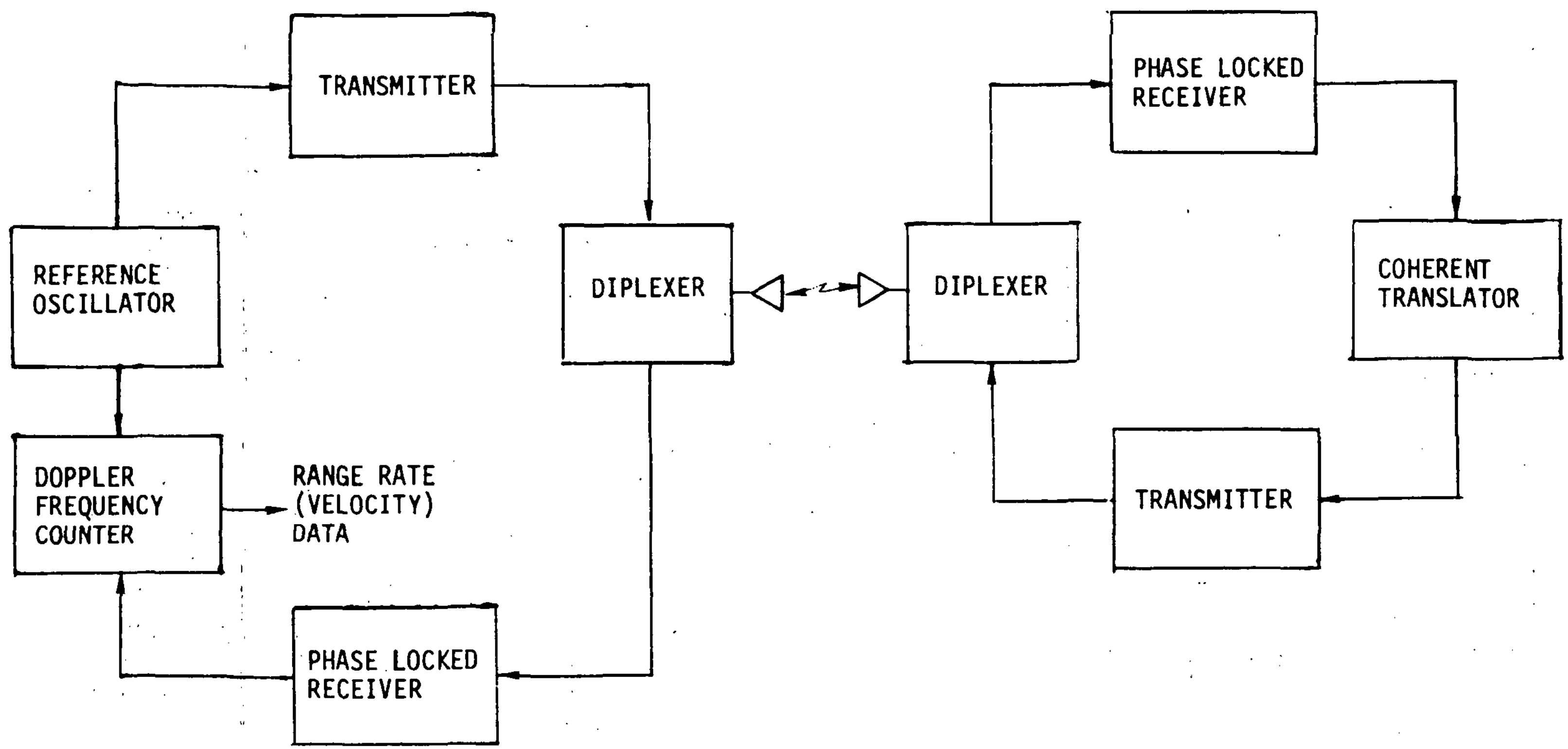

Figure 5-7. Range Rate (Velocity) Measurement Mechanization

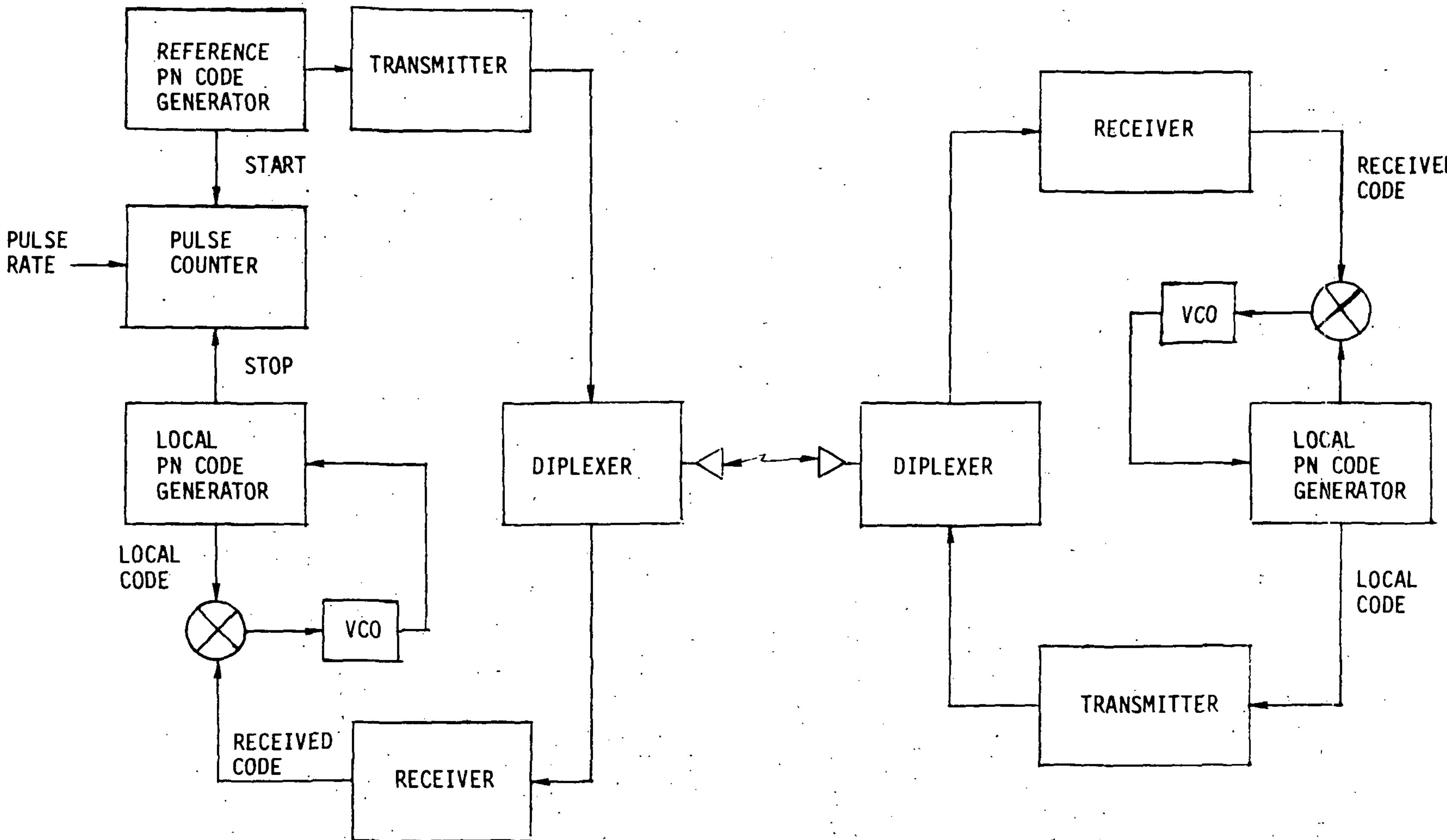

Figure 5-8. PN Code Range Measurement Mechanization

The RF and doppler measurement portion of the receiver is essentially the same as that for the tone system. The primary difference is in the range acquisition circuitry. Digital techniques are generally easier to implement than analog techniques, so the digital range measurement mechanization should be easier to implement. However, an easier implementation does not necessarily mean that a digital code system will be less costly than an analog tone system.

System Design Evolution

The long lifetime of the space station requires that the onboard systems be flexible and easily maintained. Modularized subsystems and components would allow the system to grow and change as the operational objectives change and would permit incorporation of applicable new technology as it becomes available. An evolutionary system design would initially employ existing designs and hardware and progress methodically and sensibly toward a long-term system configuration.

At first, the tracking and communication functions most likely will consist of or be an outgrowth of the Apollo Unified S-Band (USB) system, primarily operating with the MSFN ground stations. As the program evolves, the primary mode might be implemented using the TDRS, with a movement toward higher frequencies (e.g., Ku-Band) for the data relay; as TDRS becomes operational, reduced ground operating costs would tend to suggest a limited MSFN capability, functioning primarily in a backup mode.

During the growth process, the data processing, digitization, removal of data redundancy, and error correction coding might be added or increased in scope and the emphasis placed on an efficient, long-haul link through the relay satellite. As all-digital links are introduced, the use of time division multiplexing (TDM) and other signal spectrum simplifications would tend to preclude the use of analog tone ranging systems in favor of digital code systems. As discussed in a later section, considerable communication efficiency can be realized by combining the data and ranging functions into a composite digital signal.

Furthermore, recent advances in high-speed switching devices and micro-electronics, particularly large-scale integration (LSI), have provided the tools necessary to implement high data rate, all digital communication links for future operational programs. When weight, power, and reliability constraints in a spacecraft environment are considered, digital communication (plus tracking) can offer a number of significant advantages, not only in the quality of the data for a given signal bandwidth and effective radiated power (ERP), but also in the potential operational efficiency that can be achieved through automatic data handling. Flexible data formats, automatic mode control, and realtime computer processing are among the capabilities greatly facilitated by a digital transmission system. Thus, digital RF links can be expected to be the primary candidates for use on the Space Station and Tug. This should involve direct phase modulation of the data on the carrier and TDM of the various data sources.

Reference 2 recommends the use of a K-Band TDRS link as the primary means of space-to-ground communications, with omnidirectional coverage at lower frequencies for links to the logistics (shuttle) vehicle, detached experiments, and EVA. Except for an emergency voice service, provided by a separate analog system, all links and the services they carry use digital modulation, time division multiplexing (TDM), and associated source and channel encoding. Such a configuration was selected to 1) provide the maximum communication capabilities that technology would permit in the mission time frame and 2) avoid the problems and incompatibilities encountered in the Apollo USB communication system. This design philosophy provides a greater degree of independence between the functions of ranging, antenna tracking, and data transmission.

System Size, Reliability, Maintenance, and Cost

Reference 1 discusses candidate tracking systems for use on the Space Shuttle. For those systems which might also become candidates for the Space Station or Tug, the expected spaceborne transponder would occupy on the order of 1 cu. ft., weigh on the order of 20 to 50 lb., and have an MTBF of 2,000 to 10,000 hours. Since the candidates in Reference 1 apparently do not consider shared facilities among the tracking and communication services, that portion of the composite transponder devoted to ranging functions and not also associated with communication services might represent only about 10 to 20% of the total. Thus, the spaceborne equipment specifically attributable to the ranging function could conceivably occupy on the order of 0.1 to 0.2 cu. ft. and weigh on the order of 2 to 10 lb. A similar separation of the total MTBF results in an MTBF for the circuits unique to the ranging function on the order of 12,000 to 110,000 hours.

As noted in Reference 3, effective reliability and quality assurance cannot by themselves provide the long life assurance required for the five to ten year Space Station mission. The equipment life objectives would also require using the crew and resupply missions for maintenance and providing spares. This would require suitable onboard checkout equipment to carry out the maintenance program plus a maintenance philosophy such as

1. Repair will be accomplished only by replacement, either manually or by switched redundancy, based on established priority.

2. Fault isolation will be to the lowest replaceable unit level.

3. Repair below the lowest replaceable unit level will be by experiment only.

Each subsystem would have to provide suitable monitor points and possibly built-in test equipment or self-test functions. At one extreme, the crew would perform the necessary test sequences, while at the other, an automatic checkout system (operating in conjunction with the onboard computer system) would continuously monitor and determine performance parameters.

Since the expected reliability of elements within the communication system is relatively low compared to the mission lifetime, redundancy should be provided by parallel equipment, such as multiple transmitters and receivers, modulators, and demodulators, multiplexers, etc. Configuration switches would be provided to select different combinations of information channels, receiver-transmitter pairs, and antennas, to provide sufficient redundant paths to help ensure fulfillment of the full set of communications requirements.

Reference 2 suggests the use of three-level redundancy where feasible, providing an operational spare in addition to the parallel units required during antenna handover. The antennas would not have this level of redundancy due to space and weight requirements, but the availability of more than one antenna, and the availability of more than one RF channel*, would provide multiple backup modes of operation.

Cost data for a transponder alone (like specific mechanizations) are difficult to obtain because the transponder is part of a larger system. Reference 1 provides the following cost data for several communication and ranging systems.

APOLLO VHF RANGING (RCA)

VHF transceiver (CM)	\$500,000 single unit
Digital range generator (CM) Range tone transfer assembly (LM)	\$500,000 single unit

* Reference 2 suggests a K-Band section (with three steerable antennas for uninterrupted relay satellite handover with a standard spacecraft mission attitude) to handle all communications traffic but emergency voice between the Station and the ground, an S-Band section for telemetry and wideband communications with the Shuttle (ALS) and detached experiments module (EM_1), and a VHF section for dedicated, emergency voice and data between the Station and ALS or to the ground, either directly or via TDRS.

VHF R AND R SYSTEM (RCA)

	$ Million
Development	
Production	2.5
8 sets of VHF dual transceivers	1.6
16 sets of ranging units	1.2

VHF RENDEZVOUS R AND R SYSTEM (MOTOROLA)

	Non-Recurring Cost $ Million	Recurring Cost $ Million
Qualified Ranging Units	0.2	
Master unit		0.03 each
Remote unit		0.015 each
VHF transceiver	0.1	0.015 each

APOLLO S-BAND RENDEZVOUS SYSTEM (MOTOROLA)

	Non-Recurring Cost $ Million	Recurring Cost $ Million
Engineering model ranging unit	0.15	0.03
Engineering model transponder (Modified)	0.3	0.08
Antenna with pedestal, space qualified	0.8	0.14

TDRS MECHANIZATION

As noted in Reference 4, the basic navigation meausrements provided via TDRS will be 1) total range from ground to relay plus relay to user and 2) net range rate over the link. Since the relay satellite is essentially stationary relative to the ground station(s), the range from the relay to the user can be easily determined, and the measured range rate is effectively that between the relay and user.

For the purpose of this discussion, the TDRS system will be assumed to operate in a "bent-pipe" mode (References 5 and 6), where the TDRS provides only frequency translation and amplification of signals processed in either direction through the relay. The ranging signals would be generated on the ground, transmitted to a TDRS, and relayed to the user on a different carrier frequency. The user spacecraft would act as the transponder, receiving, processing, and retransmitting the processed signal on yet another carrier frequency. The TDRS would receive and relay the processed signal to the ground on still a fourth carrier frequency. The ground system would then process the returned signal to obtain the round-trip range and range rate. The TDRS orbital parameters themselves will be assumed to be determined using a separate ranging system operating on a channel used exclusively by the TDRS for maintaining its own status.

Some of the factors which influence the selection of a particular mechanization for the Space Station and Tug navigation system for use with TDRS are mission design, equipment multifunction useage, and communication link parameters (carrier frequency, signal bandwidth, transmitter power, path length, antenna gains, multipath, ionospheric and other atmospheric perturbations, etc.).

Mission Design Considerations

The Space Station is intended for low earth orbit applications requiring a minimum of ground support. The unmanned Space Tug is intended for use up to synchronous altitude and must be controlled either from the ground or the Space Station or Space Shuttle. The design of each mission would include required coverage intervals, handover procedures, etc. Many of the mission design decisions will be based on the specific mechanization selected, so several tradeoff iterations will be required in arriving at the final mission design and equipment configuration.

Equipment Multifunction Useage Considerations

While a configuration where the Space Station is the interrogator and the TDRS ground station is the transponder might improve the multifunctional aspects of the Space Station navigation system, such a configuration is unlikely. The TDRS is intended to serve a variety of manned and unmanned users, many of which having only a ranging transponder capability. So, the TDRS ground facility will of necessity contain all of the required processing equipment for determining orbital and/or trajectory parameters.

Once the Space Station parameters have been determined for a particular navigation fix, this information would be transmitted to the Station from the ground on a data channel associated with the ranging channel. The Space Station navigation system would then incorporate this updated information to improve the accuracy of its orbital calculations.

With the transponder in the Space Station, very little multifunctional use of the navigation system would be possible in terms of the Space Station obtaining ranging data from other spacecraft. If the Shuttle contained a suitable interrogator, the Shuttle could use the Station transponder, but then the Shuttle may not be able to use TDRS without also having a transponder. A possible situation would be for the Station to contain the interrogator, which has already been ruled out in terms of TDRS. A more likely configuration would be where the Station contains a separate ranging system (Space Station the active interrogator) for tracking other spacecraft. In this connection, the Station and Shuttle might operate in a manner similar to that employed in Apollo missions, where the CSM ranges to the LM using the VHF Ranging System and the LM ranges to the CSM using the Rendezvous Radar.

Communication Link Considerations

Reference 7 indicates that if TDRS is used with the Space Station, the ranging most likely will be performed using a K-Band (rather than S-Band or VHF) channel; this implies that the Station would be a high data-rate user. In order to obtain adequate communication margins over the relatively vast distances involved, steerable, directional antennas would be required both at the TDRS and at the Station. Had a low data rate been available for ranging, a VHF channel and hemispheric coverage omni-directional antennas might have been adequate.

In the case of omni antennas, operation with TDRS implies a potential multipath problem over the entire coverage interval. To combat this unwanted interference, the mechanization would be limited to a PRN digital coded system or a large-deviation (high modulation index) FM tone system. For adequate multipath protection, the code length (in time) or the period of the lowest frequency tone should exceed the two-way propagation time over the link (corresponding to a path length on the order of 50,000 n mi).

With a directional antenna on the Station, multipath becomes a problem only in the region where the Station-TDRS line-of-sight is close to grazing the earth, e.g., when the Station is on the far side of the earth prior to loss of signal (or after acquisition of signal) with the TDRS. If the mission is designed to avoid these conditions (e.g., by handing-over to another TDRS after the Station leaves the multipath zone of the second TDRS), even the fixed tone mechanization might be acceptable. Otherwise, a PRN coded or high index FM system is required. However, the coded digital system is preferred for evolution to an all-digital communication and tracking mechanization.

Since the Space Tug is designed to operate up to synchronous altitude, TDRS most likely will not provide suitable coverage. If TDRS were fitted with an omni antenna for this purpose, operation at VHF would be implied; but VHF schemes are usually fairly short-range, which would be of little use, except when the Tug were in the vicinity of the TDRS. If a longer-range capability were desired, the TDRS would require a steerable antenna with high maneuverability (e.g., 360° aximuth and 90° elevation).

TDRS Summary

The significant results of the TDRS mechanization analysis are

1. TDRS is likely to become the primary communications link between the Space Station and the ground.

2. TDRS is not a likely candidate for use with the Space Tug when it is operating at high altitudes.

3. Both the Station and Tug would contain ranging transponders, with the ranging computations performed on the ground and the results transmitted to the Station or Tug on the communication channel associated with the link used for the ranging.

4. Significant equipment commonality can be achieved between the ranging and communication services (e.g., telemetry, voice, television, etc.) if a digital format is used, but a separate system will be required for ranging to Shuttle and other spacecraft.

5. Evolution to an all-digital format makes a PRN code mechanization the preferred candidate; this provides multipath protection as well as operational efficiency; the ranging code length should exceed the two-way propagation delay (corresponding to ≈ 50,000 n mi).

6. In practice, at least one of the other candidate systems (e.g., MSFN tracking) will be available, for verification and backup modes.

MSFN MECHANIZATION

For the purpose of this discussion, MSFN, or STDN (Spacecraft Tracking and Data Network) as it is known for SKYLAB missions, can be assumed to consist of a network of unified S-Band (USB) tracking and communications stations selected from among facilities in the present MSFN, ETR, DSN, and STADAN facilities. The GRARR system at the selected STADAN sites can be assumed to have been modified to operate as a USB system. The incorporation of S-Band transponders at the selected sites is probably uneconomical, so the use of MSFN with the Space Station or Space Tug may be limited to operation similar to Apollo missions, where the transponder is located in the Station or Tug, ranging and processing is groundbased, and the resultant ephemeris data is transmitted to the Station or Tug.

As noted above, perhaps the most important mode of Space Station-to-ground intercommunications will be via TDRS. This mode can provide continuous communications with a minimum number of associated ground stations and personnel, therefore providing substantially lower operating costs compared to the present MSFN. However, MSFN is likely to be relied upon heavily during the early phases of the Space Station mission, while operational procedures and onboard capability are being verified. Furthermore, some form of long-term ground tracking capability will be required for high altitude Tug missions and for verification and backup for the primary Station system.

Mechanization Considerations

The USB system employs a PRN code ranging signal modulated on the same composite signal used for voice, telemetry, etc. For the Space Station, operating in low earth orbit, the lunar-distance PRN code is not required, allowing the use of a much shorter PRN code. The main advantage of a shorter code is shorter acquisition time. The Space Tug would require a much longer code, but it would still be significantly shorter than the lunar code.

The hybrid modulation format employed by the USB system, where various analog and digital services are frequency division multiplexed (FDM) onto the carrier, was largely dictated by the hardware complexities of onboard digital processing with, at the time the Apollo equipment was designed, the attendant penalties in weight, power, and reliability. The problems of synchronizing a variety of independent data sources and the high switching rates required to adequately sample and digitize wideband analog signals also played a prominent role in selecting hybrid analog/digital transmission techniques. Compared to an all-digital approach, these hybrid techniques require more transmitter power for a given quality (since they have lower communication efficiency), are more subject to distortion and mutual interference, and offer less operational flexibility in terms of automatic data processing and routing.

The USB system has suffered from several rather severe performance limitations and is constrained in terms of growth capability. Several of those limitations are due to improper signal design, where considerable intermodulation and incidental amplitude modulation are introduced in the antenna tracking (pointing) portion of the receiver from the subcarrier and ranging signals that are phase modulated on the carrier. While these limitations would have less affect on the Space Station, since it would not use high gain, directional S-Band antennas, they would seriously affect the Space Tug, since it would require such steerable antennas when it was operating in high altitude trajectories and orbits.

As noted above, a new system design would be needed to satisfy the overall data transmission, ranging, antenna tracking (for Tug), acquisition, and handover constraints for the entire system. The requirement is for an overall system engineering perspective in integrating the communications ranging and antenna tracking functions for an optimized subsystem conceptual design.

MSFN Summary

The significant results of the MSFN mechanization analysis are

1. As TDRS becomes operational, MSFN will probably assume a secondary role in support of the Space Station and Tug; MSFN may be prime for high altitude Tug missions.

2. Both the Station and Tug would contain transponders similar to the Apollo USB system, with ranging computations performed on the ground and the results transmitted to the Station or Tug on an associated communication channel.

3. While the USB system has a great deal of equipment commonality between the ranging and communication services, the present signal design has several shortcomings (mutual interference among services and inefficient spectrum utilization) that tend to make the USB approach unsuitable for the long-term mission; a complete restructuring of the signal spectrum and a shift to an all-digital format would be required to make MSFN competitive in function with TDRS.*

4. As with TDRS, a separate ranging system is required for tracking Shuttle and other spacecraft.

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* A great deal of emphasis will be placed on any changes incorporated in MSFN in support of the Space Shuttle program. For example, Reference 2 envisions the use of MSFN for ranging at S-Band, while TDRS is reserved for wideband digital communications.

COMBINING RANGING AND TELEMETRY SIGNALS

With a strong emphasis on the multifunctional use of the earth orbit navigation system for other mission functions, consideration should be given to the equipment simplifications possible when digital ranging and telemetry signals are suitably combined. Reference 9 studies the problem of acquiring and remaining locked to pseudo-noise (PN, or PRN) codes which have been modulated by binary data. When the code is combined modulo-2 with the data, the composite signal can convey ranging and data synchronization information in addition to the normal data information with nearly the same bandwidth and power as would otherwise be required to transmit the data alone. This represents a considerable improvement in overall performance compared to conventional digital communication techniques, such as used on Apollo.

Conventional PN code correlation techniques, such as used on the Mariner spacecraft, perform quite well so long as the data bit rate is low compared to the PN code rate (many code bits per data bit). But such schemes require many code cycles to achieve reliable acquisition and, as in Apollo, several schemes have been devised for combining several shorter codes to reduce this excessively long acquisition time. These schemes do not operate properly when the bit rate of the impressed data is nearly equal to the code rate (only a few code bits or less per data bit). This is because the resultant correlation characteristics depend more and more on the specific bit sequences in the data as the data rate approaches the code rate.

However, if a small portion of the data is allowed to carry PN synchronization information, in a proportion much less than is normally required to specify data word sync and frame sync in a conventional digital data communication system or to supply PN synchronization in a conventional ranging system, a fast acquiring digital ranging and communication system capable of operating with equal code and data rates can be used. Reference 8 discusses this "super sync" approach, and shows how it can achieve very short acquisition times, even when the channel error rate is high (e.g., $\simeq 10^{-2}$ bit error rate).

TRANSPONDER SYSTEM MECHANIZATION

The "two way" ground located navigation aid systems are characterized by ground transponders transmitting RF signals to an orbiting spacecraft in response to signals sent by the spacecraft to the ground transponders. Both range and range rate information can be obtained onboard the spacecraft. Figure 5-9 illustrates the ground transponder system concept for orbital navigation.

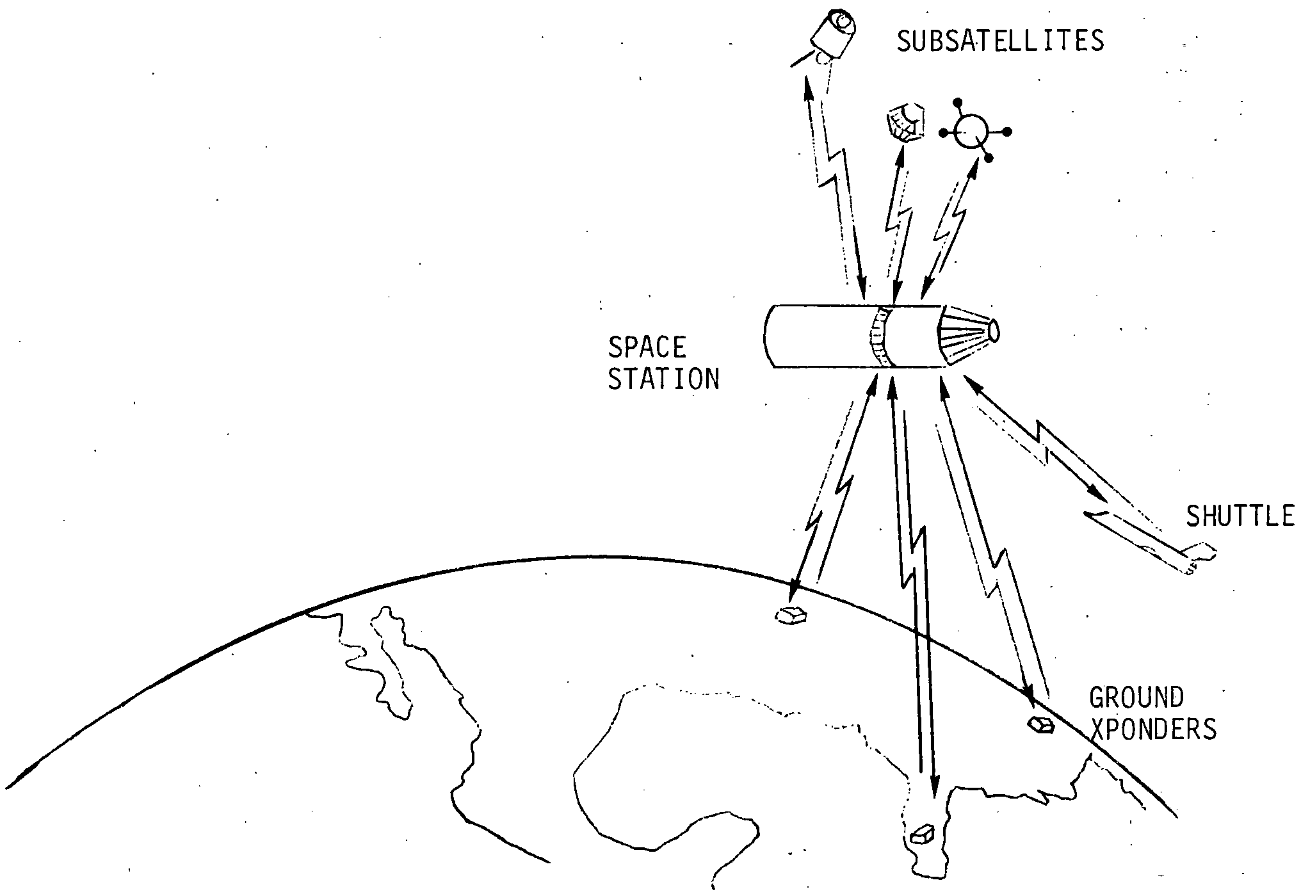


Figure 5-9. Ground Transponder System Concept for Orbital Navigation

The basic system mechanization considered here consists of an interrogator (transmitter/receiver) onboard a spacecraft orbiting the earth and a set of ground based transponders (receiver/transmitter) placed at specified locations on the earth's surface. The interrogator transmits an S-band RF signal that is received by the selected transponder and then retransmitted. The return signal received by the interrogator is processed onboard the spacecraft to obtain a measurement of range and range rate relative to the ground transponder. The range data is obtained by measuring the phase shift of a range modulation signal imposed upon an RF carrier while the range rate data is obtained by measurement of the carrier Doppler shift.

Currently there are two interrogator-ground transponder CW range/range rate systems which have undergone a considerable amount of study and analysis and have potential application as earth-orbital navigation systems. They are the Cubic Corp. CR-100-4 System and the Motorola AROD (Advanced Range and Orbit Determination) System. Both systems are similar in many respects but do differ significantly. They will be described below.

Cubic CR-100-4

The CR-100-4 concept consists of an airborne interrogator that sequentially interrogates ground transponders to measure slant range and range rate relative to the transponders. The CR-100-4 CW range/range rate system determines range by measuring phase differences of modulation frequencies and range rate by measuring carrier doppler shift. Figure 5-10 illustrates the CR-100-4 range/range rate measurement mechanization (Reference 1).

The CR-100-4 is an upgraded version of the extensively flight tested CR-100-1 system and has been designed to meet or exceed Shuttle requirements. The performance goal for the CR-100-4 system is to provide accurate range and range rate information during all phases of Space Shuttle operations including ascent tracking, orbital navigation update, orbital rendezvous, re-entry navigation update, terminal approach and landing. Cubic Corp. claims that range and range-rate measurements meeting or exceeding the general requirements of the Space Shuttle can be achieved with only relatively minor electrical modification to the existing CR-100-1 design (Reference 9).

Table 5-1 is a comparison summary and lists the important parameters and differences between the basic CR-100-1 and the upgraded CR-100-4 systems (Reference 9). Item 1 of Table 5-1 lists each contribution to the 18.5 dB signal improvement required in the coherent carrier loop to expand the maximum tracking range to 1500 nautical miles. Additional specifications for the CR-100-4 system are listed in Table 5-2 (Reference 1).

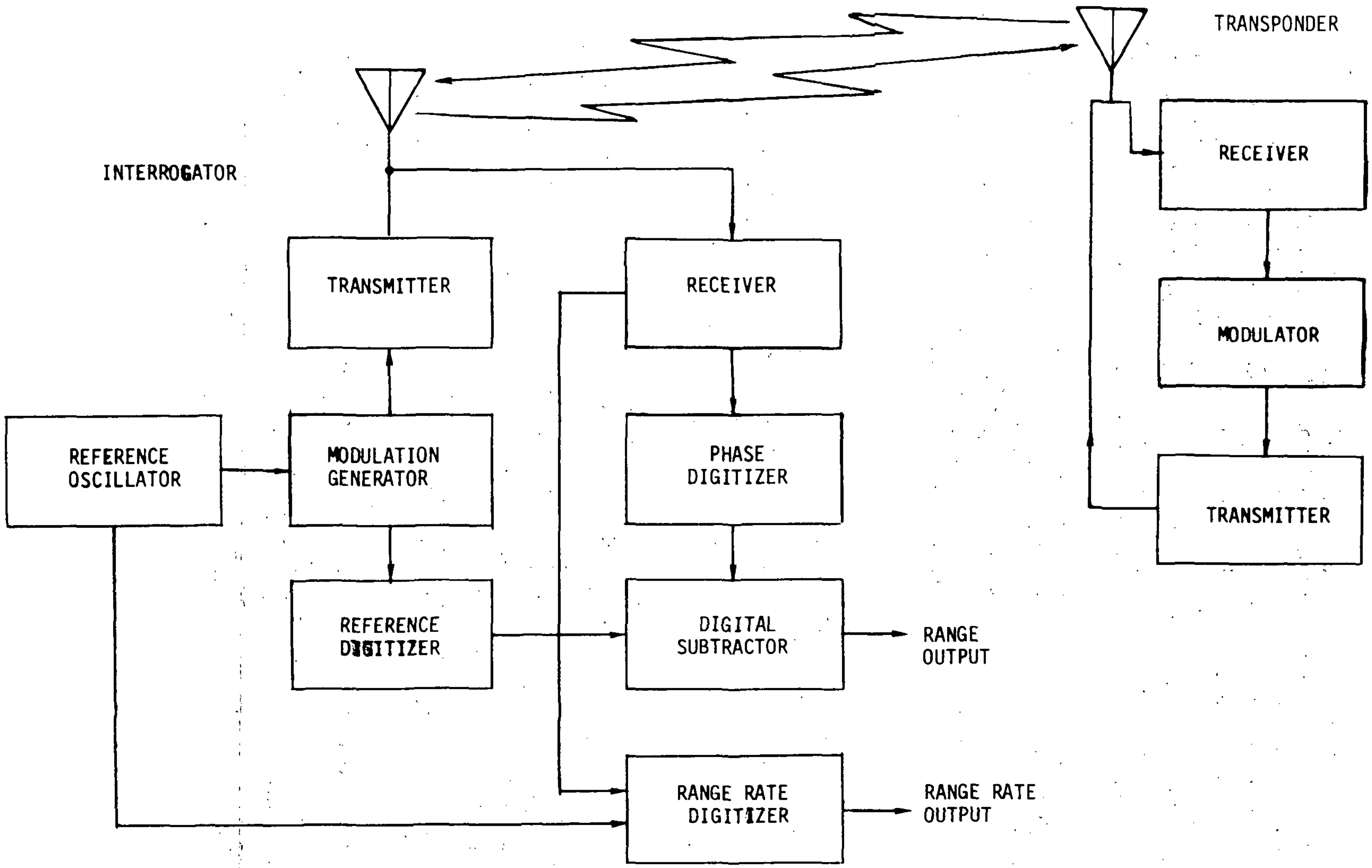


Figure 5-10. CR-100-4 Range/Range Rate Measurement Mechanization

Table 5-1

Comparison Summary, CR-100-1 CIRIS Vs. CR-100-4 (Space Shuttle)

Subsystem Characteristic	CR-100-1 CIRIS Model	CR-100-4 Space Shuttle	Space Shuttle Design Improvement (See 1A-1F) Gain	Tracking Range
1. Maximum Range	200 miles	1500 n.miles		175 n.miles
A. Transmitter Power	4 watts	12 watts (solid state)	+5 dB	312 n.miles
B. Receiver Noise BW	6 kHz	1 kHz	+8 dB	784 n.miles
C. Data Loop Noise	All Tones	All Tones or 1 Tone Sequential	+5 dB	1400 n.miles
D. Noise Figure	7 dB	5 dB	+2 dB	1760 n.miles
E. Antenna Cable Budget	24 ft (-2 dB)	40 ft (-1 dB)	+1 dB	2000 n.miles
F. Frequency	1.6 GHz	2.2 GHz	-2.5 dB	1500 n.miles
			18.5 dB	
2. Signal Fade Margin				
A. Transmitter Power Module	- - - -	60 watts	+7 dB	From 12 watts.
	- - - -			
B. Ground Antenna	- - - -	5 dB gain at 10° elevation angle	+2 dB	3 dB has already been allocated on the signal power budget.
			+9 dB	
3. Tracking Range	200 miles-200 ft	1500 n.miles-3 ft	Requires a one step change in transmitted power. Since the transmitter power must be switched, the equipment can track as close in as 3 ft. and not exceed receiver dynamic range.	

Table 5-1

Comparison Summary, CR-100-1 CIRIS Vs. CR-100-4 (Space Shuttle) (Continued)

Subsystem Characteristic	CR-100-1 CIRIS Model	CR-100-4 Space Shuttle	Space Shuttle Design Improvement
4. Dynamics			
A. Velocity	5000 ft/sec	25,000 ft/sec	(1) Modify sweep acquisition, (2) Widen range rate servo loop, (3) Increase $\dot{R}$ counters from 20 bits to 22 bits.
B. Acceleration	1000 ft/sec^2	1000 ft/sec^2	Requires no change.
5. Range Modulation Tones			
A. Number of Tones	4	5	Extends unambiguous range to 1400 n. miles.
B. Modes	All tones simultaneous	Simultaneous and sequential	Sequential reduces data loop noise and extends tracking range.
6. Sweep rate	One rate	Two rates	(1) Slow rate orbital. (2) Fast rate normal. Selectable by data link.
7. Transmitter	One power level	Two power levels	Low power level for close in range. Selectable by data link.
8. Data Link			
A. Acquisition	8-bit word	8-bit word	Word repeats until system acquisition is complete.
B. No. of Addressable Transponders	255	63	Additional capacity is allocated for transmit and sweep functions.
C. Bit Rate	2500 bits/sec	1000 bits/sec	Improve SNR at increased range.

Table 5-1
Comparison Summary, CR-100-1 CIRIS Vs. CR-100-4 (Space Shuttle) (Continued)

Subsystem Characteristic	CR-100-1 CIRIS Model	CR-100-4 Space Shuttle	Space Shuttle Design Improvement
9. Range Rate Servo	BW = 200 Hz	BW = 500 Hz	Reduce acquisition time in orbital mode.
10. Range Servo			
A. Fine	BW = 40 Hz	BW = 20 Hz	Improve SNR.
B. Coarser Tones	BW = 40 Hz	BW = 50 Hz	Decreases settling time for each servo.
11. Range Master Oscillator	Stability 1PP 10^6	Stability 1PP 10^7	Reduces scale factor error at maximum range.

Table 5-2

Additional Specifications for CR-100-4

Volume	
Interrogator	0.41 ft^3
Transponder	0.73 ft^3
Weight	
Interrogator	25 lbs
Transponder	20 lbs
Carrier Frequency	2.2 GHz
Modulation Frequencies	54 Hz - 240 kHz
Receiver Sensitivity	-118 dBm
Power Consumption	
Interrogator	125 W
Transponder	100 W
MTBF	
Interrogator	2000 hrs
Transponder	6500 hrs
Acquisition Time (Maximum)	5.4 sec
Dynamic Range	80 dB
Measurement Rate	0.25/sec Modified 20/sec
Antennas	Omnidirectional Throughout

The CR-100-4 system basically consists of an airborne (or space-borne) interrogator set, which interfaces with the on-board computer, together with transponders that are located on the ground. The spaceborne interrogator sequentially and selectively addresses ground transponders, and determines slant range to the transponder by measuring the round trip time delay of five tones phase modulated onto the carrier. The carrier frequencies (2) are identical for all transponders and are in the 1 to 2 GHz band. Therefore, only one interrogator can communicate with only one ground transponder during an interrogation time period. The phase angle of the highest tone modulation determines the resolution in the time delay measurement. The lowest tone establishes the maximum range increment of the system and the three in-between tones permit the ambiguities to be successively resolved. The five measured phase angles are converted into an unambiguous range in the on-board computer. The maximum unambiguous range is approximately 1400 n.mi. using the five tones. To minimize multipath interference, the modulation index of the fine tone phase modulation is about 20, a value requiring a large receiver bandwidth and relatively large transmitter power for a given range.

The range rate measurement is based on tracking of the received carrier frequency by a digitally controlled frequency synthesizer. The difference between the transmitted and received frequency (Doppler frequency) is a measure of range rate. Upon reception at the interrogator unit, a suitable multiple of the two-way Doppler frequency is extracted and counted over a given time interval. The resulting count is proportional to range rate.

Desired transponders are selected by using a subcarrier for the transmission of an address code and transponder configuration code. The subcarrier, along with the range tones, phase modulates the carrier. During periods of no ranging, voice or data may be transmitted to and from the transponder through baseband modulation of the carrier. Ground transponders are continuously receiving, but will retransmit signals only after their proper identification code has been received. Up to 63 transponders can be selectively called. The airborne navigation computer selects a transponder of known coordinates by serially transmitting the address code to the interrogator unit. After detecting the address, the transponder replies with its own code. The navigation computer then verifies that the response is by the proper transponder before it accepts range or range-rate data. A typical interrogation cycle takes one second (References 10, 11).

Motorola AROD

The Advanced Range and Orbit Determination System is an S-band, CW radio tracking and navigation system which provides accurate information on the position and velocity of a space vehicle. This information is derived from measurements of slant range and range rate between the vehicle and several transponder stations. Measurement, data collection, and position determination can be performed on-board the vehicle in near-real-time. Simple ground transponder stations, controlled by the vehicle, may be placed in remote unattended locations without the need for inter-station communications.

Range is measured by the time delay of the propagated radio wave. The range modulation is a pseudo-noise (PN) code which bi-phase modulates the downlink S-band carrier. Range rate is determined by the Doppler frequency shift of the carrier. Figures 5-11 and 5-12 illustrate the AROD range and range rate measurement mechanizations, respectively (Reference 12). Table 5-3 provides specifications for the AROD system (Reference 1).

Line-of-sight ranging and thus the measurement of range and range rate can be maintained with up to four ground transponders simultaneously, providing a continuous readout of range and range rate to the four sites. The orbitor radiates a modulated carrier which is received by all ground transponders within range. The selected transponders coherently translate the received signal to the appropriate channel frequency to provide adequate frequency isolation between the four transponders. They then transmit the new carrier frequency containing the reconstructed range modulation. The four channel receiver in the vehicle separates the received signals to determine the slant range and radial velocity to each transponder (Reference 10).

A one-way VHF link is used to control the acquisition and operation of the ground transponder stations. A time division multiplexed signal is transmitted which sequentially controls the four transponder stations in use. The VHF message contains information pertaining to the site identification being called, the S-band channel frequency on which the site is to respond and the control data. In addition, the VHF link performs three other functions:

1) Provides steering information for the narrow beam S-band antenna from a VHF direction finding system (if an omni antenna is not used)

2) Presets the S-band receiver to the proper frequency based upon the Doppler shift contained on the VHF signal and

3) Presets the PN code to the approximate correct position by a time marker contained in the VHF data.

The above three functions minimize the amount of time to acquire a new station by reducing the search of space, frequency, and time, respectively.

The orbiting spacecraft S-band transmitter radiates a biphase modulated signal at 2200 MHz which is received by all the transponder stations within range. The S-band signal is acquired by the transponder phase-locked re-

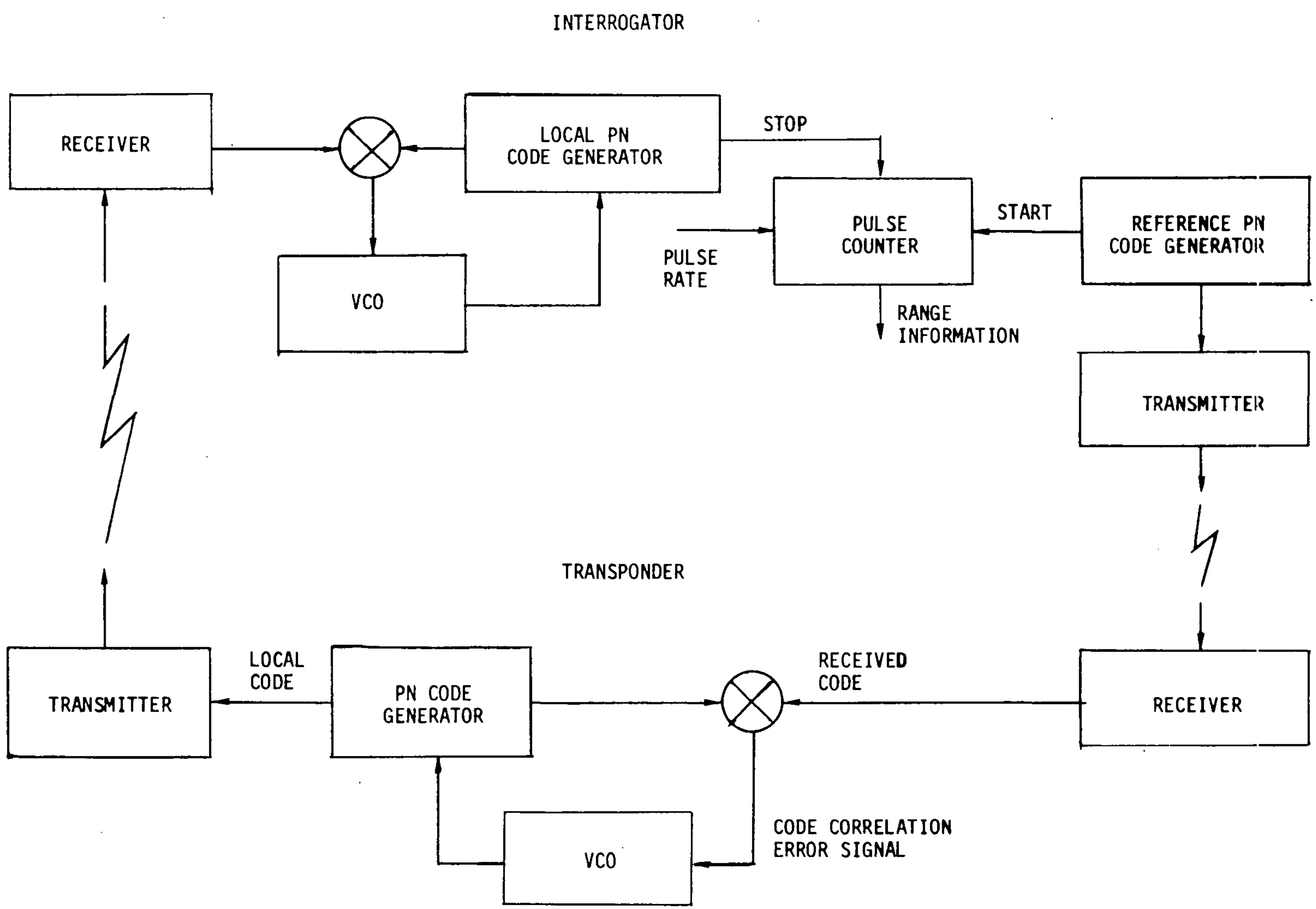


Figure 5-11. AROD Range Measurement Mechanization

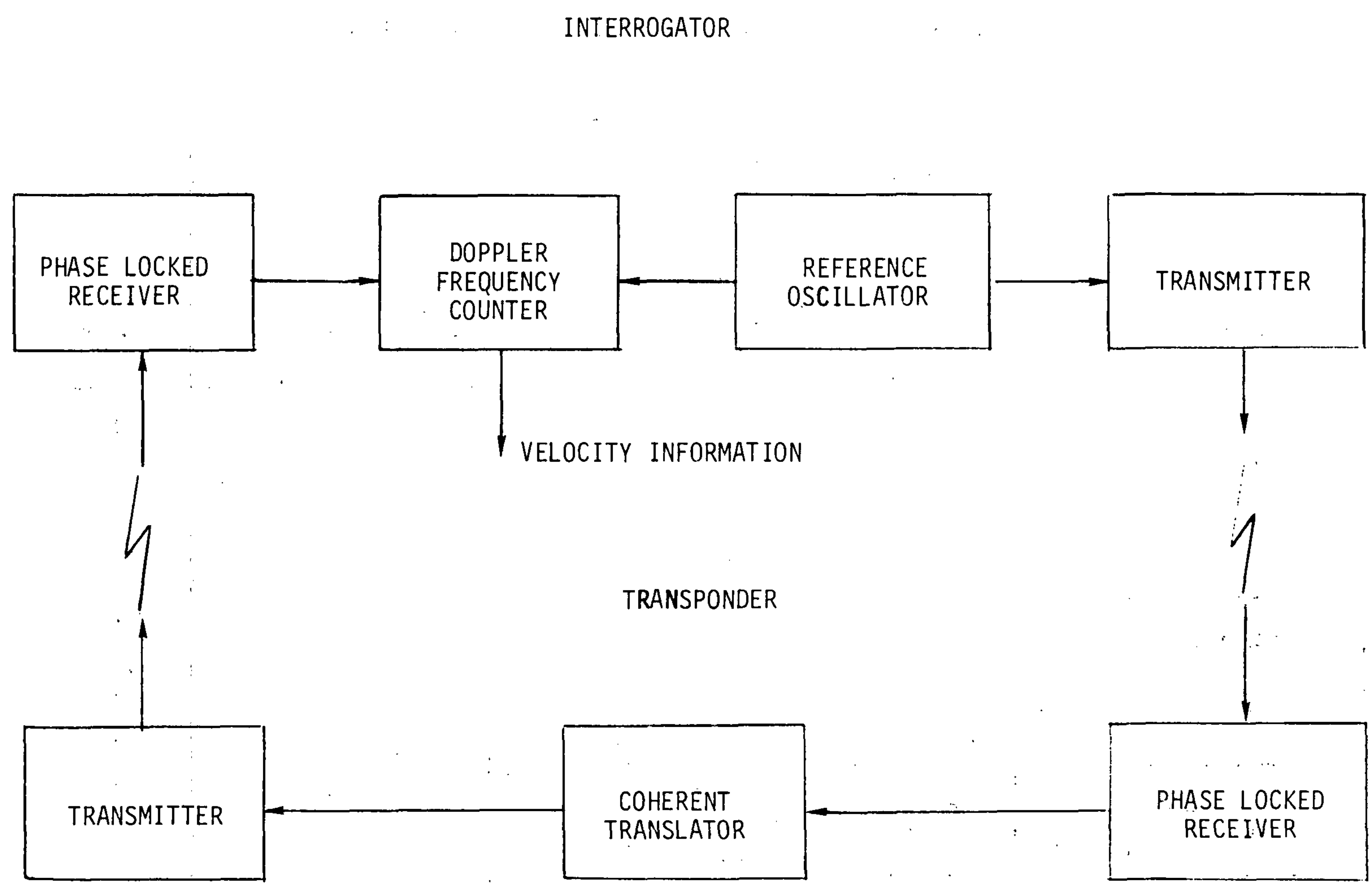


Figure 5-12. AROD Range Rate Measurement Mechanization

Table 5-3

AROD Specifications

	Interrogator	Transponder
S-BAND		
Transmitter power	10W	10W
Antenna gain	3 dB	16 dB
Frequency band	2200-2450 MHz in 400 kHz steps	1750-1850 MHz in 400 kHz steps
Receiver threshold	-127 dBm	-127 dBm
Data rate	50 bits/sec	50 bits/sec
VHF		
Transmitter power	6W	-
Antenna gain	3 dB	3 dB
Frequency band	135-150 MHz in 25 kHz steps	135-150 MHz in 25 kHz steps
Receiver threshold	-	-126 dBm
Data rate	400 bits/sec	400 bits/sec
VOLUME	0.94 ft^3	12 ft^3
WEIGHT	50 lb	260 lb
PRIME POWER	150W	120W
MTBF	2000 hr	Undetermined
[ADDITIONAL SYSTEM SPECIFICATIONS]:		
ACQUISITION TIME	6.5 sec	
DYNAMIC RANGE	66 dB	
MEASUREMENT RATE	4 times/sec	
RANGE		
Maximum (threshold)	10,800 n.mi.	
Maximum (unambiguous)	1,642 n.mi.	
Minimum	0	
RANGE RATE		
Maximum	$\pm$ 44,300 ft/sec	
Minimum	0	

ceiver where time correlation of the PN code is achieved. The carrier is reconstructed, coherently translated to an uplink frequency in the 1800 MHz band, and is bi-phase modulated with the local PN code. The first signal returned by the transponder S-band transmitter contains a reverse Doppler shift so that the frequency search at the vehicle is minimized (Reference 10).

The four-channel S-band receiver in the vehicle phase locks to the received signal establishing full coherence over the two-way link, which allows the extraction of range and range rate information. The complete set of data is read out four times each second for processing by an on-board computer and for transmission via a separate telemetry set if desired.

Operational Usage

Selection of the ground transponder locations must be based upon the orbits under consideration, which depends upon the particular mission, and the desired navigation accuracy. The primary factor in determining an acceptable set of station locations is the longest period of non-coverage that can be allowed before the navigation error growth exceeds requirements. Other considerations that influence the choice of station locations are:

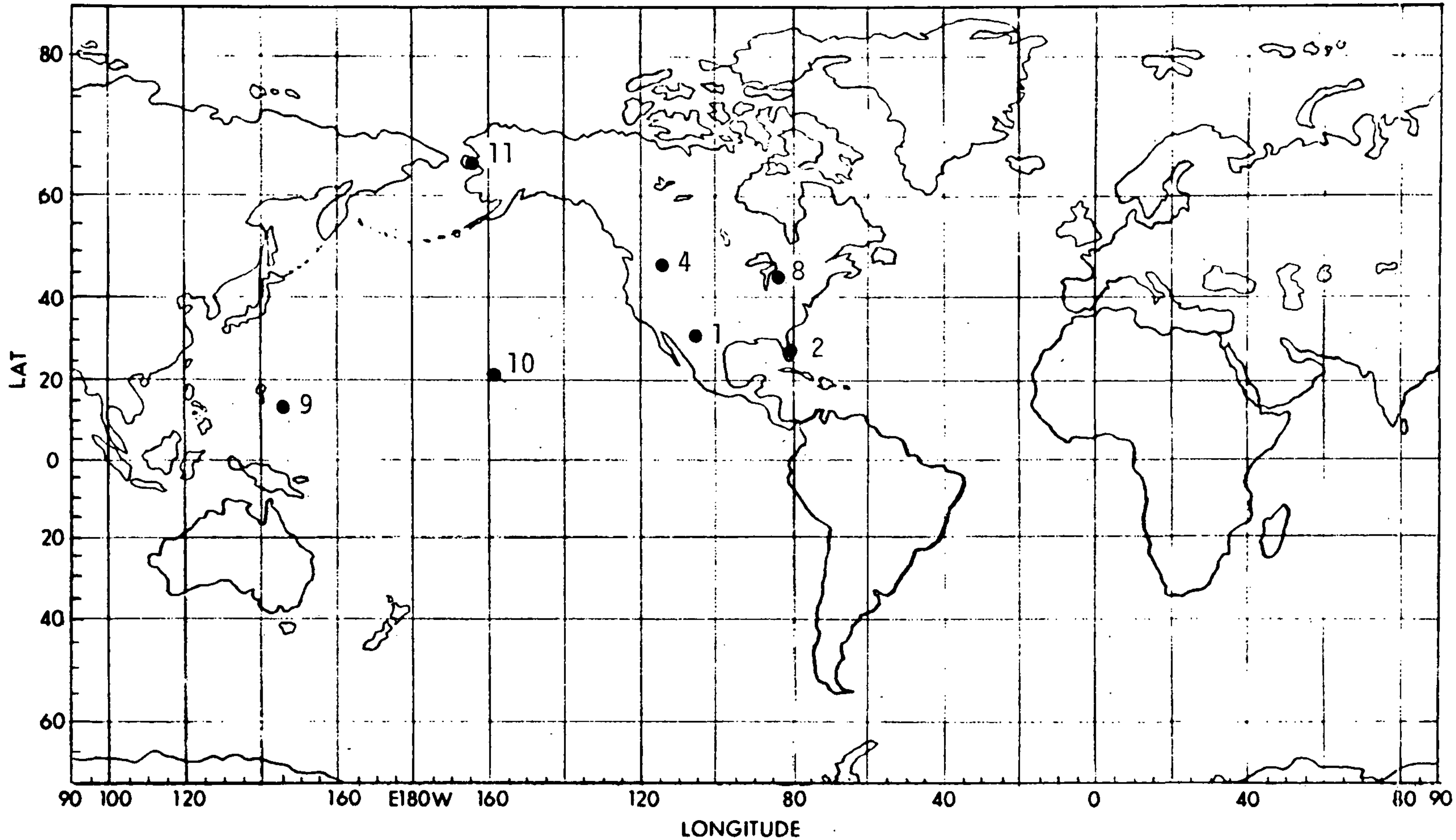
1) the number of stations required should be minimized to reduce the cost of installation and maintenance;

2) the location of all stations within U.S. territory is highly desirable;

3) the location of stations at operating airfields or military bases is highly desirable to facilitate maintenance.

A candidate set of beacon locations obtained from Reference 4 is presented in Figure 5-13. The seven stations are located in U.S. territory and provide adequate coverage except for extremely low inclination orbits of 5 degrees or less (Reference 4). To adequately cover these low inclination orbits it is necessary to have transponder locations in foreign countries. For a 270 n.mi. circular orbit at 55 degrees inclination, the maximum interval of non-coverage is approximately 2.5 revolutions. For a 200 n.mi. orbit at 90 degrees inclination, the maximum interval of non-coverage is approximately 1.75 revolutions.

Transponder redundancy can be achieved by installing additional transponders throughout the ground track area or by installing redundant transponders at each location. All ground transponders are similar except each responds to its own location or address code. The transponder receivers are always operating. However, the transponder transmitters operated only when called upon by means of the location code.



1. Biggs AFB, Texas (31.85° N. Lat., 106.38° W. Long.)
2. Cape Kennedy (28.65° N., 80.57° W.)
4. Fairchild AFB, Washington (47.8° N., 117.5° W.)
8. Wurtsmith AFB, Michigan (44.5° N. Lat., 83° W. Long.)
9. Anderson AFB, Guam (13.58° N., 144.92° E.)
10. Honolulu Intl. (21.33° N., 157.92° W.)
11. Alaska (65° N., 165° W.)

Figure 5-13. A Candidate Set of Ground Beacon Locations

Low Orbit

For the ground transponder systems considered, coverage is limited only by line-of-sight conditions. Thus, for a nominal mission circular orbit of 270 n.mi., the maximum line-of-sight range is approximately 1400 n.mi. The ground range subtended for this radio range is approximately 1300 n.mi. That is, the spacecraft is visible if the "subsatellite" point is within 1300 n.mi. of the transponder station (Reference 13). Both transponder systems considered have been designed for a maximum tracking range of approximately 1400-1600 n.mi. Therefore, spacecraft operation in a 200-280 nautical mile circular orbit is completely covered by either the AROD or CR-100-4 systems. Figure 5-14 plots the tracking time for elevation angles above five degrees for various orbit altitudes as a function of the angle between the transponder position vector and the orbit plane. As can be seen, the available tracking time is between ten and twenty minutes for 100 nautical mile altitude orbits up to the range limiting altitude.

Synchronous Orbit

In order to support a space tug at an altitude on the order of 20,000 n.mi., a ground transponder system is required which has a maximum threshold range in excess of 20,000 n.mi. It is also desirable that the maximum unambiguous range be greater than 20,000 nautical miles; although the range ambiguity could possibly be resolved by the onboard computer. At the present time, neither the AROD nor the CR-100-4 system has such a capability. Both systems need considerable upgrading to support the space tug during synchronous orbit mission phases (additional discussion under Paragraph 1.6).

Multifunction Mechanization Considerations

Two of the factors which influence the selection of a particular mechanization for the space station and tug navigation systems for use with a ground located transponder system are the multifunction mechanization aspects of the navigation and communication links.

Navigation

The Motorola AROD and Cubic Corp. CR-100-4 systems are strictly navigation range and range rate measurement systems. Both can accurately supply navigation data to earth-orbiting spacecraft up to a maximum range of approximately 1400-1600 n.mi. All that is required in the space station is an interrogator and S-band omni antenna. In addition, ranging to other spacecraft is possible providing the target vehicle possesses a transponder.

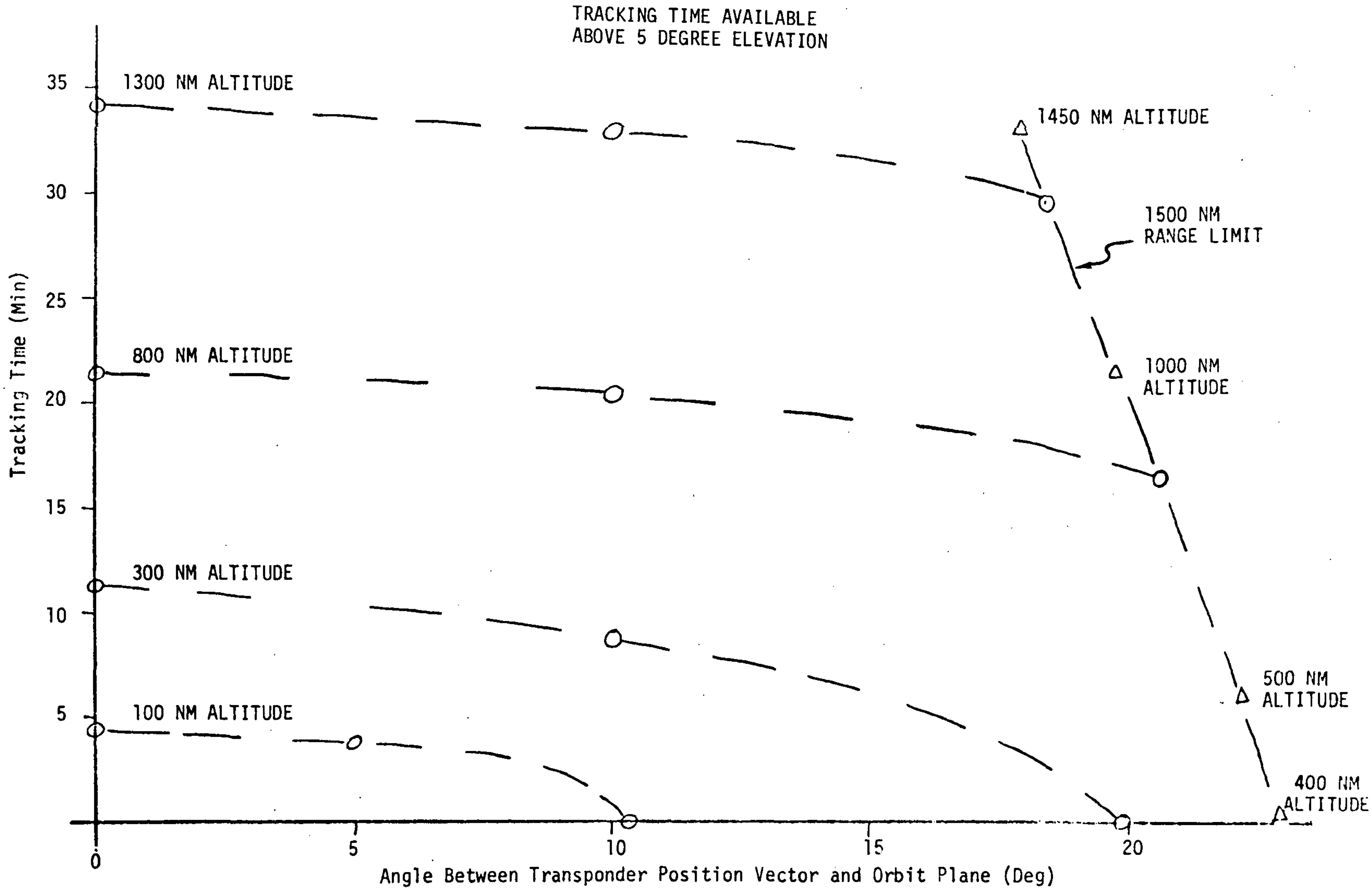


Figure 5-14. Tracking Time for Various Orbit Altitudes

This means that cooperative rendezvous can be supported with any other spacecraft which has a transponder. Since neither transponder system possesses an angle measuring capability, an optical tracker is required to obtain angle data during rendezvous. Another alternative would be to have a gimbaled antenna on the space station. However, the dollar cost of such a gimbaled narrow beam antenna would be high, even if an "off the shelf" antenna were available. For instance, the cost of building another space qualified LM rendezvous radar antenna system (gimbaled) would be in the neighborhood of $500,000 (single item cost). The weight of the LM rendezvous radar antenna subassembly is approximately 44 pounds and has an MTBF of 2,000 hours. The total weight of the rendezvous radar system is approximately 78 pounds. The gimbaled antenna with a single cubic interrogator would weigh about 69 pounds.

Communication

Due to the remote unattended nature of the transponder ground station locations, neither AROD nor the CR-100-4 system contain any significant communication capabilities. Any communications capability desired which is independent of the ranging function, either to ground transponders or to other spacecraft, will have to be supplied by separate communications equipment.

There is the possibility that both transponder systems could be modified to include a communication capability. Baseband voice or data modulation could be transmitted to and from the transponder when no range measurements are being made. However, ground transmission links would have to be developed.

Error Model

This section presents error budgets for the measurement of range and range rate during orbital mission phases for two different ground transponder systems.

Cubic CR-100-4

The range error budget for the maximum range case (1500 n.mi.) is given in Table 5-4 and lists error sources for both the range random error and the range bias error (Reference 9).

Table 5-4

CR-100-4 Range Error Budget

I. RANDOM ERROR

Error Source	1σ Magnitude
A. Ranging error due to finite signal-to-noise ratio and equipment added noise	1.0 ft, 0.3 m
B. Phase shift uncertainty over dynamic range of ranging operations	1.0 ft, 0.3 m
C. Phase shift with temperature over operating environment	1.0 ft, 0.3 m
D. Phase shift of interrogator due to vibration, shock and g-loading	negligible
E. System error due to craft dynamics (25,000 ft/sec and 1,000 ft/sec^2)	0.2 ft, 0.06 m
F. Multipath error in ground-to-air range links	3.0 ft, 0.9 m
G. Digitization Error	0.3 ft, 0.09 m
RSS TOTAL	3.5 ft, 1.05 m

II. BIAS ERROR

A. Calibration (Equipment)	1.0 ft, 0.3m
B. Scale Factor	
Stability of crystal oscillators	0.1 ppm

The range rate error budget is given in Table 5-5. For both the low and synchronous orbit phases, where the spacecraft is far above the atmosphere, the range rate error is essentially determined by the stability of the crystal oscillator and the uncertainty in the velocity of light in space (Reference 9).

Motorola AROD

The range error budget is shown in Table 5-6 (Reference 1). The overall range error is mainly determined by multipath and propagation error contributions. The multipath error of 6m (worst case) is reported to be an off-the-cuff estimate by Motorola since no experimental data exists. It might be that the multipath errors measured for the Cubic System might also apply to AROD.

Table 5-6 also gives the AROD range rate error budget. As for the Cubic system, for both low and synchronous orbits, the range rate error is essentially determined by the stability of the crystal oscillator and the uncertainty of the speed of light in space.

The range and range rate accuracies listed represent the errors resulting from the equipment implementation. The bias errors are those due to uncompensated open loop equipment delays which vary due to temperature changes, circuit aging, and similar effects. The errors due to noise effects are the result of having a finite signal-to-noise ratio in the system and, as range increases, the accuracy is downgraded. These accuracies represent those which are obtained for a single measurement. Continuous, repeated measurements provide a smoothing effect on the data which in some cases can improve the accuracy.

Errors due to propagation effects are of an extremely complex nature. By use of the most sophisticated correction techniques available and under the best propagation conditions, the errors introduced by propagation effects are comparable to the equipment errors. Under poorer propagation paths and with less sophisticated correction techniques, the propagation errors would probably be an order of magnitude greater than the equipment errors.

A study of the effect of tropospheric refraction on radar range and range rate measurements (Reference 14) has been made on an object at orbital height (250 n.mi.). The results show that for elevation angles greater than five degrees the small effect of surface refractivity fluctuations suggests that onboard corrections to range and range rate can be made by fitting simple functions to the data. This may be done provided the orbital height is fixed. Such a procedure should result in errors no greater than 6 meters in range and .1 meter per second in range rate.

Table 5-5

CR-100-4 Range Rate Error Budget

I. VELOCITY INDEPENDENT

	Error Source	1σ Magnitude
A.	Rate error due to finite signal-to-noise ratio and equipment added noise	.01 ft/sec, .003 m/sec
B.	System error due to craft dynamics, a = 1,000 ft/sec^2	.001 ft/sec, .0003 m/sec
C.	Digitization Error	.014 ft/sec, .004 m/sec
D.	Multipath	.01 ft/sec, .003 m/sec
	RSS TOTAL	.02 ft/sec, .006 m/sec

II. VELOCITY DEPENDENT

Stability of Crystal Oscillator	1 ppm

Table 5-6

AROD Error Budget

<u>RANGE</u>	
Resolution	0.183 m
Bias (calibration, stability of crystal oscillator, uncertainty in speed of light)	$\pm$ 0.5 m (1σ)
Noise (2000 km)	0.2 m (1σ)
Noise (20,000 km)	1.25 m (1σ)
Multipath	6 m (worst case)
<u>RANGE RATE</u>	
Resolution	0.026 m/sec
Bias (stability of crystal oscillator)	1.5 ppm
Noise (2000 km)	0.015 m/sec (1σ)
Noise (20,000 km)	0.045 m/sec (1σ)
Multipath	Undetermined

In addition to measurement errors caused by equipment accuracy and propagation effects, the accuracy to which the vehicle position and velocity can be determined is also affected by the accuracy of the transponder coordinates. Multiple passes over transponder sites can assist in more accurately determining the transponder location.

Cubic CR-100-4 Cost Data

The listed cost figures (Reference 15) refer to equipment which has been qualified and is in a state of final configuration, including all necessary documentation and tested in accordance with MIL-E-5400 and MIL-4158:

Non-recurring Cost	$1,700,000
Spaceborne Interrogator	150,000 - each
Ground Transponder	60,000 - each
Ground Support Equipment	250,000

If a flyable prototype is required (hand-built with no documentation), the non-recurring cost would then be only $180,000 while the per unit cost of the interrogator and transponder would remain approximately as listed above.

It is estimated that the per unit cost of the spaceborne interrogator would go up by $20,000 if modifications for continuous and simultaneous readout of range and range data is required. Substantially higher costs for the hardware would be expected if the use of high reliability parts is a requirement.

Motorola AROD Cost Data

Similar cost figures associated with AROD (Reference 9) are:

Spaceborne Interrogator (copy of present model)	$200,000 - each
Ground Transponder (copy of present model)	75,000 - each
Ground Support Equipment	
(a) Airborne (one required)	110,000
(b) Ground (one required)	75,000
Setup for Aircraft Test (at Phoenix, Arizona)	100,000 - 150,000

The non-recurring cost of redesigning the equipment to 1971 standards (rather than 1964 state of the art techniques) is not known at this time.

Antenna Requirements

In order to achieve maximum flexibility, the S-band antenna(s) on the space vehicle (S-band so as to be compatible with the transponder systems) must have an omnidirectional pattern. This is required so that the antenna pattern covers any desired combination of transponder locations on the ground, as well as the transponder in another spacecraft, for instance, during rendezvous. Multiple S-band omnidirectional antennas may be needed on the space station in order to satisfy this requirement. Each antenna should weigh 1-2 pounds apiece.

With the AROD system, an omnidirectional VHF antenna is also required onboard the space station. The antenna pattern must cover all transponders at least during the acquisition phase. The Cubic CR-100-4 system has no such requirement.

Low Orbit

For a typical circular orbit of 270 n.mi., both ground transponder systems considered need only hemispherical ground antennas to exceed maximum range threshold requirements. From the standpoint of system complexity, cost, and reliability, it is preferable to utilize a fixed, hemispherical coverage ground antenna rather than a steerable narrow beamwidth antenna.

Synchronous Orbit

At the present time, neither the AROD nor the Cubic CR-100-4 system has the capability of tracking a space vehicle to an altitude on the order of 20,000 n.mi. To accomplish such a feat would require at a minimum, for both systems, an increase in the maximum unambiguous range, an increase in the gain of the transponder antenna, and an increase in the power output of the interrogator and transponder transmitters. The SNR improvement obtained by the upgrading of the CR-100-1 (indicated in Item 1 of Table 1-1) to the CR-100-4 is representative of the additional improvement required to operate at synchronous orbit. The required additional gain (about 22 dB) probably cannot be obtained without a redesign of the system.

The use of a directional S-band antenna at a transponder site is an undesirable feature since it complicates the equipment required, resulting in a less reliable system. While the directivity of the antenna helps to reduce the amplitude of multipath signals, it also increases the probability of multipath due to the additional antenna height required above ground.

Due to the undesirable complexity and cost associated with the choice of a steerable transponder antenna (a direction finding system is needed to steer the antenna), which is inconsistent with the philosophy of a simple, easily maintainable transponder set-up, coupled with the increased power and unambiguous range requirements, it is felt that the ground transponder system is an unattractive candidate to supply navigational data to a space tug during synchronous orbit mission phases.

Reliability

Because of the ten year life of the space station it is imperative that the system mechanization for the ground transponder system include sufficient redundancy and inflight maintenance to provide satisfactory operational failure detection and reliability.

The mean time before failure (MTBF) of the CR-100-4 interrogator is expected to be in excess of 2,000 hours and the MTBF of each ground transponder is over 6,500 hours as predicted in Reference 1. However, Reference 10, a more preliminary document, predicts MTBF's of 9,300 hours and 24,200 hours for the interrogator and transponder if screened parts are used. If one transponder per ground station and one interrogator were the operational configuration, then to support a ten year mission would require many maintenance calls to replace failed equipment. The redundancy required is a functio n of the increased reliability desired versus the increase in overall dollar cost.

Consider a system with four CR-100-4 interrogator units, each totally independent of the others and controlled by the spacecraft computer system. Failure decisions would be made by the computer rather than the interrogators themselves. Any failure mode detectors within the system would also be considered to be redundant.

Consider that two transponders are available at each site. Any one combination of any one transponder at a site and any one of the four interrogators available will provide complete system operational success.

It is assumed that no more than one interrogator and one transponder are active at any moment. The remaining transponder and the three remaining interrogators are inactive unless selected as replacements.

All possible combinations of the four interrogators and two transponders that provide system operation are indicated in Figure 5-15. The resulting system MTBF is 4,950 hours or 6.9 months. Additional reliability is possible with increased redundancy, but at a greatly increased dollar cost. In order to support a ten year space station mission, an inflight maintenance schedule is required whereby failed units will be removed and replaced. In the event of an interrogator failure, one of the three remaining interrogators will be selected as a replacement. In the event of a transponder failure, the remaining transponder will become active.

The MTBF for four interrogators in parallel (three inactive) is 8,000 hours or 11.1 months. Removal and replacement of the failed interrogators is made easy by the compact, modular design of the units. The MTBF for two transponders in parallel is 13,000 hours or 18.1 months.

While the above discussion made use of the Cubic CR-100-4 system MTBF figures, the same arguments hold true for the AROD system. The estimated MTBF for the Motorola AROD interrogator likewise has been given as 2,000 hours (Reference 1). This could be achieved using the latest components and packaging techniques. No estimate exists for the ground transponders. However, they will be all solid state and should have a MTBF higher than the space-borne unit since they are much simpler in design.

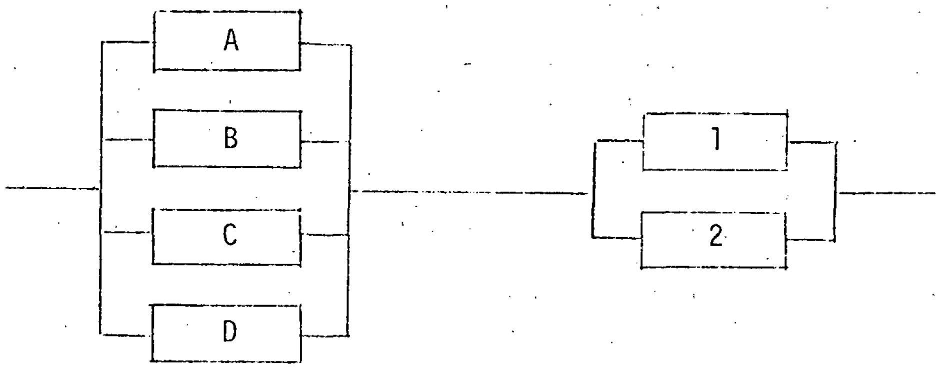


Figure 5-15. Reliability Diagram

Increased reliability could also be obtained through the use of high reliability parts (References 9 and 10), but at an increased cost. Moreover, use of high reliability parts probably would be cheaper than using redundancy because of the high cost per unit.

Potential Problem Areas

For low earth orbits, both the AROD and CR-100-4 systems possess the capability of furnishing accurate range and range rate information to an orbiting space vehicle. Both systems, however, have relatively low MTBF figures when a mission of 10 year duration is contemplated. To increase unit reliability, the equipment design should be as simple as possible and the latest high-reliability components and packaging techniques should be utilized. Interrogator and transponder redundancy should be employed to increase system reliability.

Neither the AROD nor CR-100-4 system is capable of supporting the space tug at a synchronous altitude of approximately 20,000 n.mi. The AROD system is limited to a 1,642 n.mi. range, while the CR-100-4 is limited to a 1,400 n.mi. range. Significant equipment changes would be required for both in order to support the space tug mission. To accomplish this objective the minimum following steps are required:

- Increase the maximum unambiguous range to 20,000 nautical miles.
- Increase the interrogator and transponder output powers.
- Increase the gain of the ground transponder antennas.

The disadvantage in increasing the gain of the ground transponder antenna in order to support space tug mission phases is the resultant complexity and reduced reliability associated with a directional S-band antenna. In addition, an antenna pointing system must be added.

The fact that the AROD system requires a VHF link is also undesirable because of the added complexity and the need for an extra antenna on the space vehicle.

Ground Transponder Summary

A ground transponder system provides an effective navigation means for a low earth-orbiting spacecraft. First, one radio navigation system can serve both orbital and atmospheric mission phases. Second, the state of the art in two-way ranging techniques, exemplified by Motorola's AROD and Cubic Corporation's CR-100-4, indicates little development risk. Further, all indications are that the transponder units can be made easily maintainable and portable and certainly less costly to operate as compared to MSFN/TDRS and other ground tracking techniques.

At the present time the AROD and CR-100-4 systems are relatively comparable when considering such factors as range, accuracy, multipath, acquisition time, interference susceptibility, dynamic range, operation and maintenance. However, the CR-100-4 system shows a significant advantage over the AROD system when considering the factors of complexity/reliability, operational experience, and to a lesser extent, cost.

The AROD system design and circuitry is more sophisticated than for the CR-100-4 system. This increased complexity may lead to a lower MTBF figure for the AROD system, assuming both systems use the same type of components and packaging techniques. The AROD system also requires an additional VHF link which further increases complexity, thus further reducing the system MTBF.

An additional point is that extensive field experience has been gained in the operation of the CR-100-4 through the operational use of its predecessors, notably the SECOR, SHIRAN, and CR-100-3 systems. The AROD system has no such experience.

As a result of the above comparisons, at the present time the CR-100-4 system is considered to be a significantly more attractive candidate for a ground transponder system than the AROD system.

Both ground transponder systems considered are presently unsuitable for synchronous orbit navigation. Each system needs extensive upgrading to support such orbital mission phases.

HORIZON SENSOR SYSTEM

Several possible approaches to the mechanization of a star and horizon sensing orbit navigation system for the NASA Space Station and Tug vehicles have been considered. In formulating the mechanizations, the following Space Station requirements were assumed:

1) The Space Station will not be maneuvered to provide scanning for horizon or star measurements.

2) Functional redundancy sufficient to provide automatic failure detection is required for both sensing functions.

3) The orbit navigation accuracy provided should be limited only by the earth's horizon pehonmenology, i.e., not by the onboard sensors.

4) The inertial attitude reference mechanization will consist of a gyro reference package with star sighting updates used to bound error growth.

The operational altitude of the Space Station is limited to approximately 300 nautical miles. However, the Tug is required to operate out to synchronous altitude. In the case of the Tug, some attitude maneuvering for sensor sightings is considered allowable.

Star Sensing Approaches

Three basic approaches to star sensing exist; gimballed trackers, strapdown (electronically gimballed) trackers, and star mappers (slit scanners). This last category requires some spacecraft rotation to provide star transits across a sensor slit and consequently is not suitable for the Space Station. For a candidate gimballed star tracker, the design developed for the NASA/TRW Precision Attitude Determination System (PADS) was selected because of its 3 $\widehat{\text{sec}}$ accuracy capability (Reference 47).

Current strapdown tracker design approaches use an image dissector tube to provide electronic scanning of a fixed field-of-view. The field-of-view is determined by the optical system used and must be selected as a compromise between accuracy and the size of the field-of-view. The START design proposed by TRW for the Space Shuttle was chosen as a representative strapdown tracker (Reference 48).

The strapdown tracker will require a sun shade whose size and weight are dependent upon the smallest angle between the sensor and the sun (or earth) for which the sensor must operate. This can be a serious problem since shades 4 to 6 feet long are required to get within 20° of the sun or earth. The definition of the shade requirements is dependent on the sensor location on the vehicle, the vehicle attitude profile, orbit characteristics, etc. The matter becomes further complicated if the star tracker is also used as a rendezvous sensor since the relative position of the sun, earth and rendezvous target must also be considered. For the purposes of this tradeoff, the sun shade for the strapdown trackers was not included since the requirements are not sufficiently defined.

Horizon Sensors

Horizon sensors have been used extensively in spacecraft attitude control systems. Consequently, there is a wide variety of design approaches available. In order to satisfy the Space Station accuracy goal, the edge tracking or scan through approaches are most suitable. Two current designs are the Quantic Mod IV system (Reference 49) and the TRW ATS system (Reference 50). The Quantic Mod IV was selected for this tradeoff because of its advantages in reliability and altitude range capability. This system can be used from low altitude orbits all the way out to synchronous altitudes. The system design includes four scanning sensor heads which provide the functional redundancy necessary to detect the failure of a sensor head.

An alternate horizon sensing approach using an electronically scanned sensor is also considered in this tradeoff. The NASA/MSC Multi-Mode Optical Sensor (MMOS) uses an image dissector tube to electronically scan across the horizon (References 51 and 52).

An S-20 detector is used which requires an ultra-violet radiance profile mechanization rather than the infrared (CO_2) band usually employed. This sensing approach is limited to measurements of the sunlit horizon which will not be significant for low earth orbit application, but will significantly affect mission planning and navigation accuracy for a high earth orbit.

In order to provide a horizon sensing mechanization with automatic failure detection capability, four MMOS would be used with their fields-of-view arranged in 90° intervals similar to the Quantic Mod IV system.

The MMOS can also serve as a star tracker although in order to use the same sensor for both star and horizon measurements, a spacecraft rotation would be required. Consequently, for the space station separate sensors will be required. A star/horizon mechanization consisting of six MMOS would provide the required functional capability for the Space Station and has been included as a candidate in the tradeoff.

Tradeoff Results

The tradeoff data for the star/horizon mechanization approaches is presented in Table 5-7 . Totals for weight, power, and size are included. The MTBF for each sensor unit is given rather than a system number in order to properly compute replacement costs. For the horizon sensor, the replaceable unit is a sensor head assembly of which there are four in the system. The estimated replacement costs are based upon the number of failures for a ten year operating life.

From the tradeoff data, there is a definite all-around advantage of the mechanization approach utilizing 2 strapdown star trackers with the Quantic Mod IV horizon sensor system. This mechanization is definitely

Table 5-7 . Star/Horizon Mechanizations

	Weight (lbs)	Power (watts)	Size (cubic inches)	Unit MTBF (hours)	Estimated Unit Cost ($)	First System Cost	Replacement Costs
2-PADS Star Trackers	78	60	5606	39,000	245K		1,100K
Quantic Mod IV	45	38	1403	256,000/head	15K/head	590K	35K
System Total	123	98	7009				1,135K
2-Strapdown Trackers*	20	16	596	25,000	45K		32K
Quantic Mod IV	45	38	1403	256,000/head	25K/head	190K	35K
System Total	65	54	1999				67K
6-MMOS*	120	72	3828	125,000	50K	300K	210K

*Without sun shade

preferable as the star/horizon navigation system for the Space Station. If attitude maneuvers are permitted, a mechanization employing 2 to 4 MMOS would be an attractive alternate approach. This mechanization is also preferable for the tug vehicle because of the problem of horizon sensing with the MMOS on the transfer trajectory.

The requirement of redundancy to provide automatic failure detection results in the use of multiple sensors. The star sensing function could be performed adequately with a single strapdown star tracker if the vehicle is in a local vertical mode or if the vehicle is maneuverable. Figure 5-16 taken from Reference 52 illustrates the probability of having at least one star in the sensor field of view, and the average number of stars in the field of view as a function of the diameter of the circular field of view. For a 15 degree diameter field of view the probability of at least one star is greater than 0.5, and the average number of stars is one. Because two stars are required for a complete alignment, at least two attitudes are required.

For space station application, three MMOS are required as a minimum - two for horizon sensing and one for star sensing. This system of three MMOS is competitive costwise with a system composed of one strapdown star tracker and the Quantic Mod IV horizon sensor package. However, the Quantic system still provides redundancy for horizon sensing and is thereby favored over the MMOS.

UNKNOWN LANDMARK TRACKING SYSTEM

A tradeoff was performed to evaluate alternate approaches to the hardware mechanization of the star-sensing and unknown-landmark tracking functions for the space station program. The system is not considered as an appropriate candidate for the tug vehicle because of its altitude limitations. Four basic candidates were configured using available technology for star-sensing mechanizations with TRW-developed unknown-landmark tracker technology:

- SPARS strapdown star sensors with the DELTRA landmark tracker
- PADS gimbaled star tracker with the DELTRA landmark tracker
- Integrated tracker using the PADS sensor assembly and the DELTRA sensor assembly mounted on the same gimbal system
- SLANT star/landmark tracker

The evaluation criteria for the mechanization tradeoff were the following:

a. Ease of calibration and alignment

b. Recurring cost

c. Weight, power, and size

d. Fields-of-view requirements

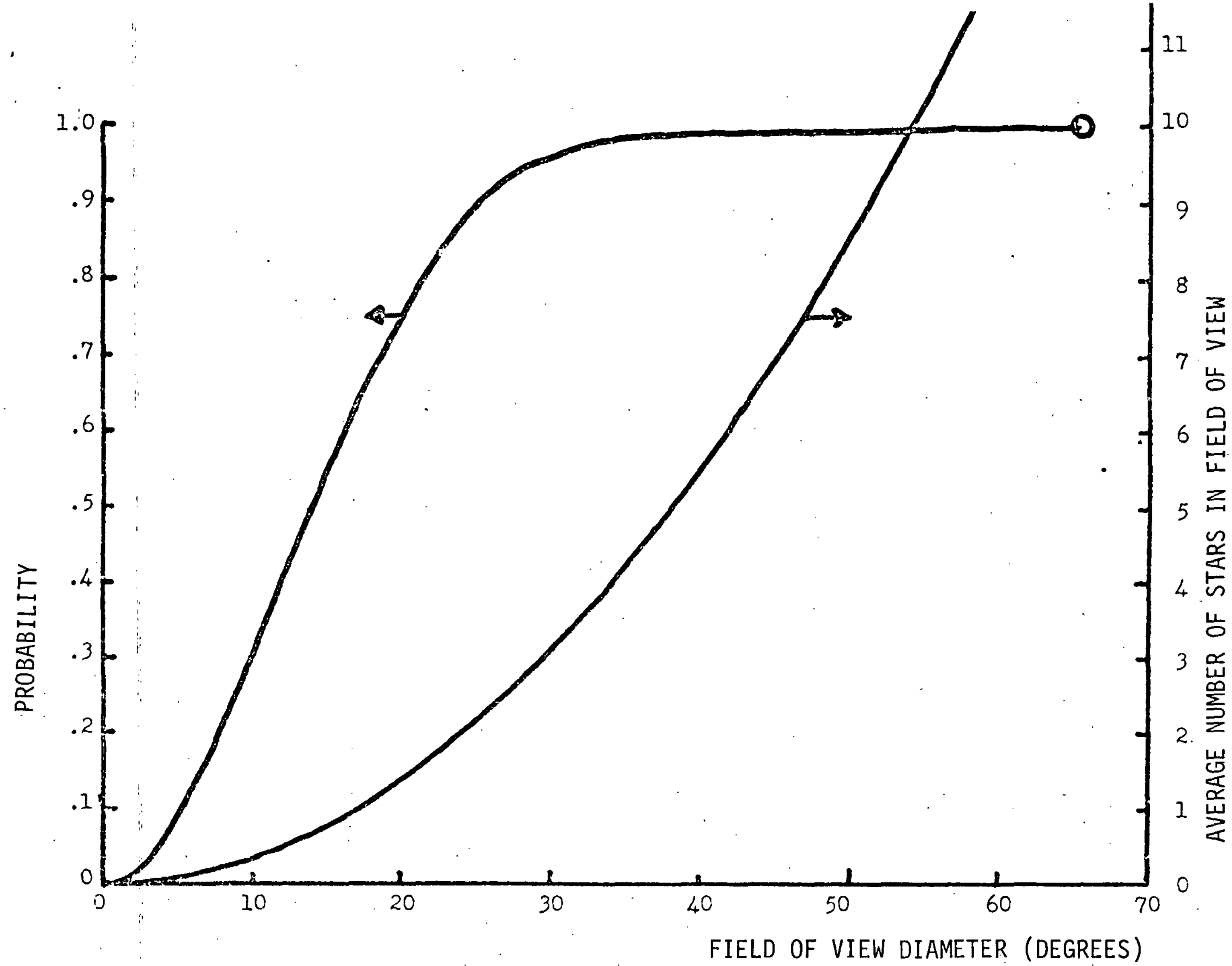


Figure 5-16. Star Availability As a Function of Field of View

e. Development status

f. Reliability

g. Orbital altitude versatility

SPARS/DELTRA Candidate

As part of Program 681D, the Air Force has developed the SPARS. This system utilizes two SSAs, which provide an output pulse whenever a star crosses one of several detector slits in each sensor assembly. The sensors each require a 4-deg field-of-view which are nominally oriented in the orbit plane looking out the top of the spacecraft with a 60-deg angle between the two sensor axes. While other orientations of the sensor axes are possible, this one has been shown to work effectively in the test programs conducted. A special interface and timing unit (SITU) is used with the SSAs to receive and process the star transit pulses to provide transit-time output data at regular intervals for input to a digital computer. Detailed descriptions of this equipment are contained in Reference 53. The projected flight weight and power for each SSA is 20 lbs and 7 w, respectively. The SITU weights 9 lbs and requires 15 w of power. Because recurring cost estimates for this equipment were not available, an estimate of $300K for two SSAs and the SITU was used for the tradeoff. Since a reliability estimate for the equipment was also not available, a value was used for the tradeoff which corresponds to the PADS gimbaled star tracker and electronics.

The SPARS strapdown star sensors rely on vehicle motion relative to the stars to provide the star transmits necessary for attitude determination. The SPARS system has demonstrated excellent performance for simulated low-altitude orbits of a spacecraft maintained in a local-vertical orientation. For higher orbital altitudes, the spacecraft rotation rate would be considerably lower, thus reducing the frequency of star transits, which should result in some performance degradation. Consequently, the versatility of this mechanization is somewhat less than that of one employing a gimbaled star tracker.

The TRW-developed DELTRA landmark tracker is comprised of 3 units in its present design form: a sensor gimbal assembly (SGU), a sensor electronics assembly (SEA), and a correlation electronics assembly (CEA). These units, along with the SPARS, SSAs, and SITU, are shown in Figure 5-17, which illustrates the SPARS/DELTRA mechanization candidate.

The DELTRA tracker requires an earth-directed field-of-view of ± 50 deg from the local vertical in the orbit plane and ± 10 deg normal to the orbit plane. Since the SPARS sensors look up and the DELTRA looks down, many potential spacecraft installations will require separation of the two sensors and some method of maintaining or monitoring relative alignment between the two, e.g., an optical alignment link as used in the PEPSY system design (Reference 53).

PADS/DELTRA Candidate

This mechanization candidate employs two gimballed trackers: one for stars and one for landmarks. The star tracker is the PADS tracker currently under development by NASA/TRW. Since the DELTRA design was developed from the PADS tracker design, many of its hardware elements are identical to the corresponding ones for PADS. This includes the gimbal system, most of the sensor assembly, and most of the SEA. As shown in Figure 5-18, this mechanization candidate is comprised of the two similar SGUs, two similar SEAs, and the CEA for DELTRA.

The field-of-view provided for the SPARS star tracker can be tailored to any specific spacecraft installation. For this mechanization, a star field-of-view of 60 deg x 30 deg situated just above the local horizontal has been selected. This field-of-view provides adequate star availability and minimizes the separation between the star field-of-view and the landmark field-of-view as illustrated in Figure 5-19. With these field-of-view requirements, the probability is higher that they can be provided from the same point in a spacecraft than if the star tracker had to look out the top. By mounting the two trackers together on the same mounting base, an alignment link between the two would not be required.

Combined Gimbal Candidate

A third mechanization candidate was configured by redesigning the PADS star tracker gimbal ring to accept the DELTRA sensor assembly (optics and detector) in addition to the PADS sensor assembly. The resulting tracker, shown in Figure 5-20, eliminates one gimbal set and one SEA from the PADS/DELTRA candidate of Figure 5-18. The gimbal system would be time-shared between star tracking and landmark tracking. The PADS system design requires star updates at 5-min intervals. By offsetting the boresight axes of the two optical systems, the slewing requirements between star tracking and landmark tracking and the required outer gimbal freedom are minimized. Consequently, the combined tracker could be devoted to landmark tracking for the majority of the time with less than one-half a minute out of every five required for a star update.

The weight, power, and cost of this tracker are significantly less than those of two separate trackers due to the elimination of one gimbal system and one SEA. The reliability is also improved as shown in Table 5-8.

SLANT

The TRW SLANT star/landmark tracker design provides for the tracking of stars and landmarks with a single gimbaled sensor. This tracker is designed to cover both the star field-of-view and the landmark field-of-view of Figure 5-19 by rotating the sensor about the outer gimbal axis. A star update is required once every 5 min that will take 1/2 to 1 min to slew up, settle, take the star update, and slew back down for landmark tracking. The tracker will consequently be available for landmark tracking 80 to 90% of the time. The complete tracker consists of a SGU, SEA, and CEA, as shown in Figure 5-21.

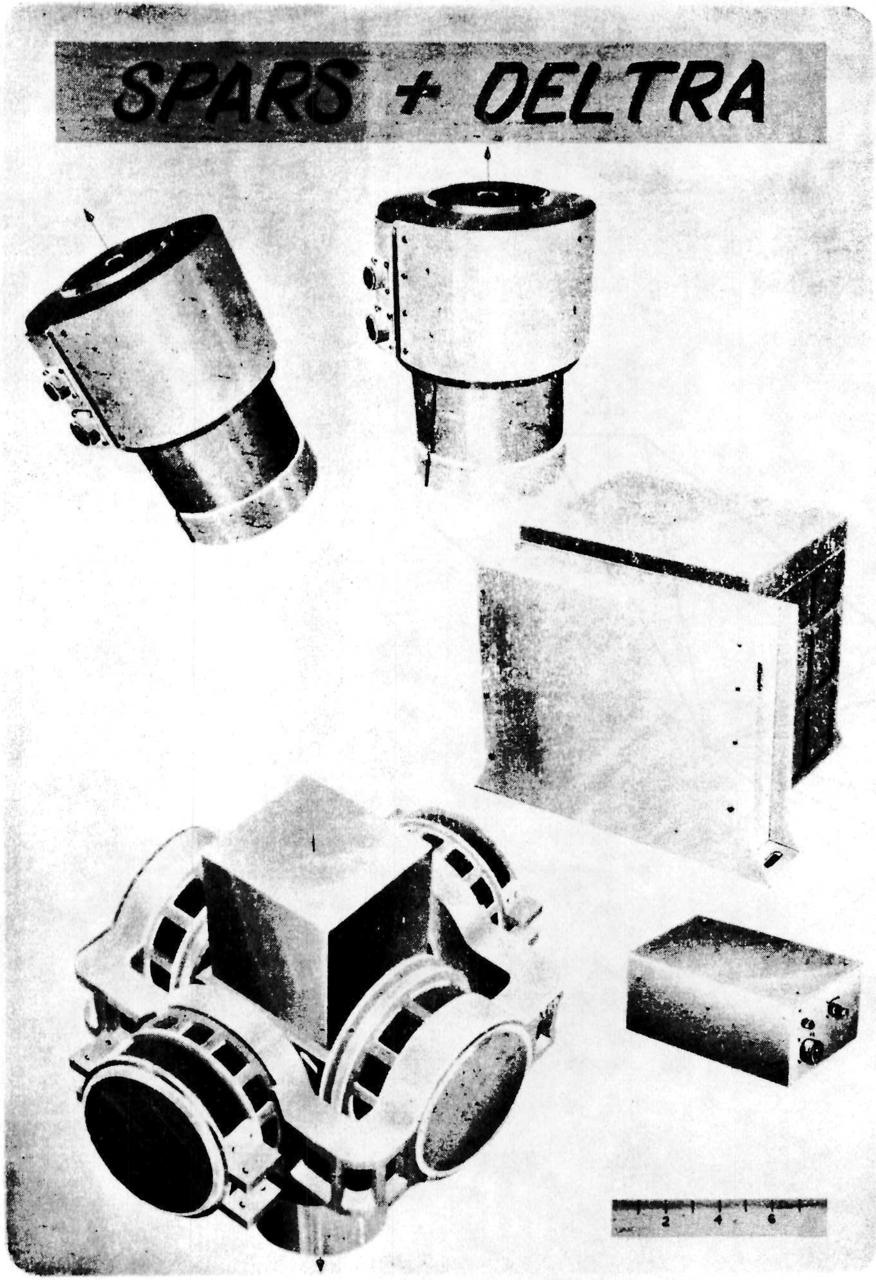


Figure 5-17. SPARS/DELTRA Candidate

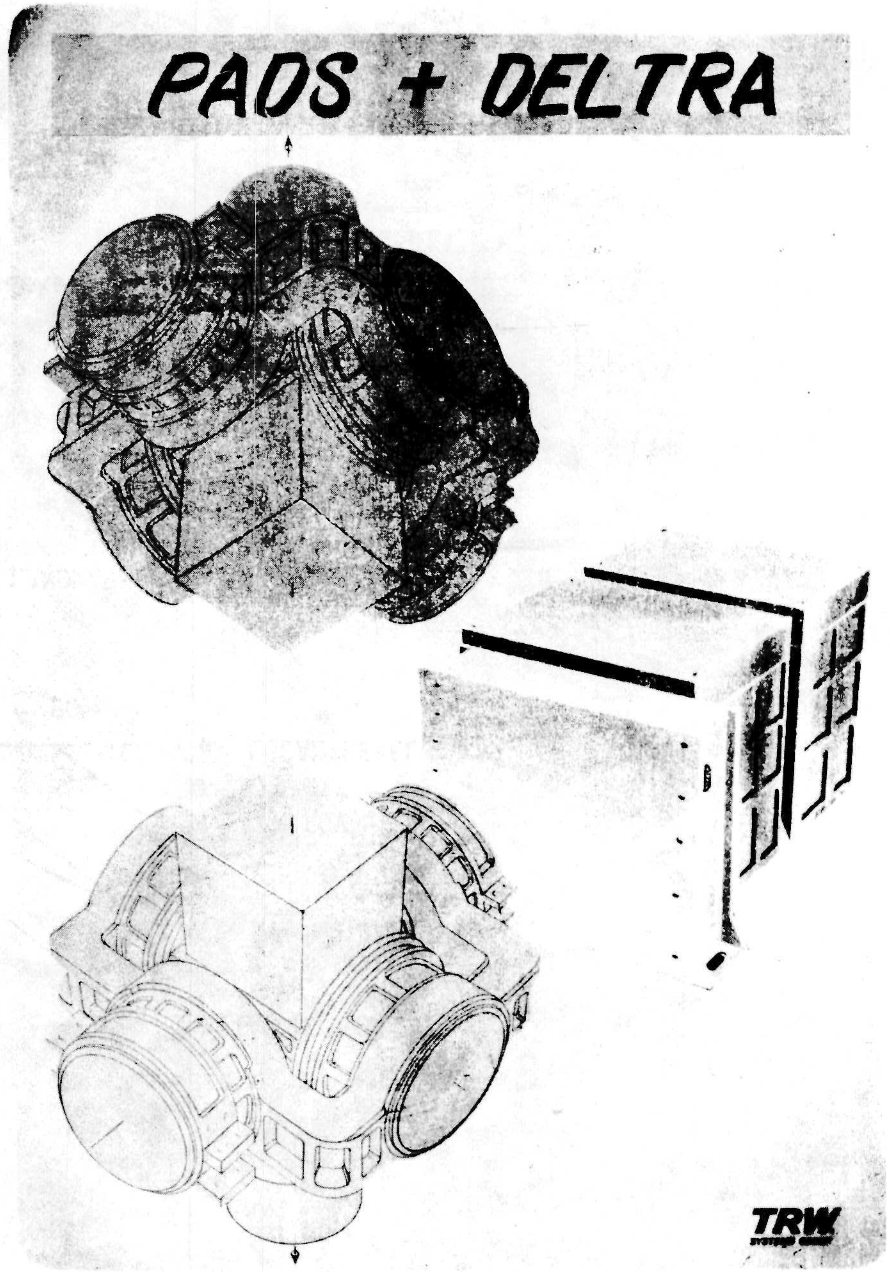


Figure 5-18. PADS/DELTRA Candidate

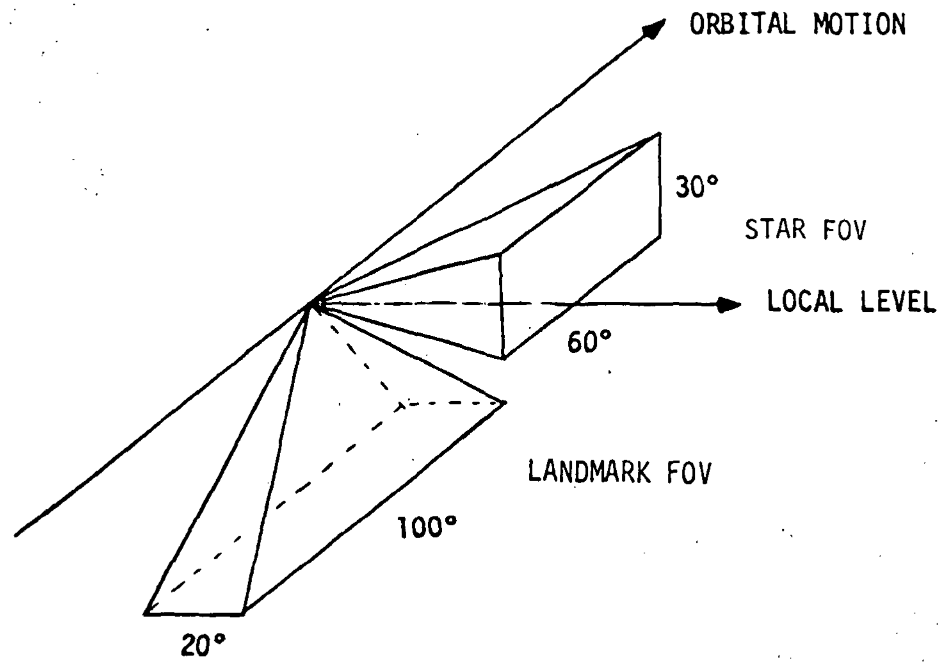


Figure 5-19. Required Tracker FOV

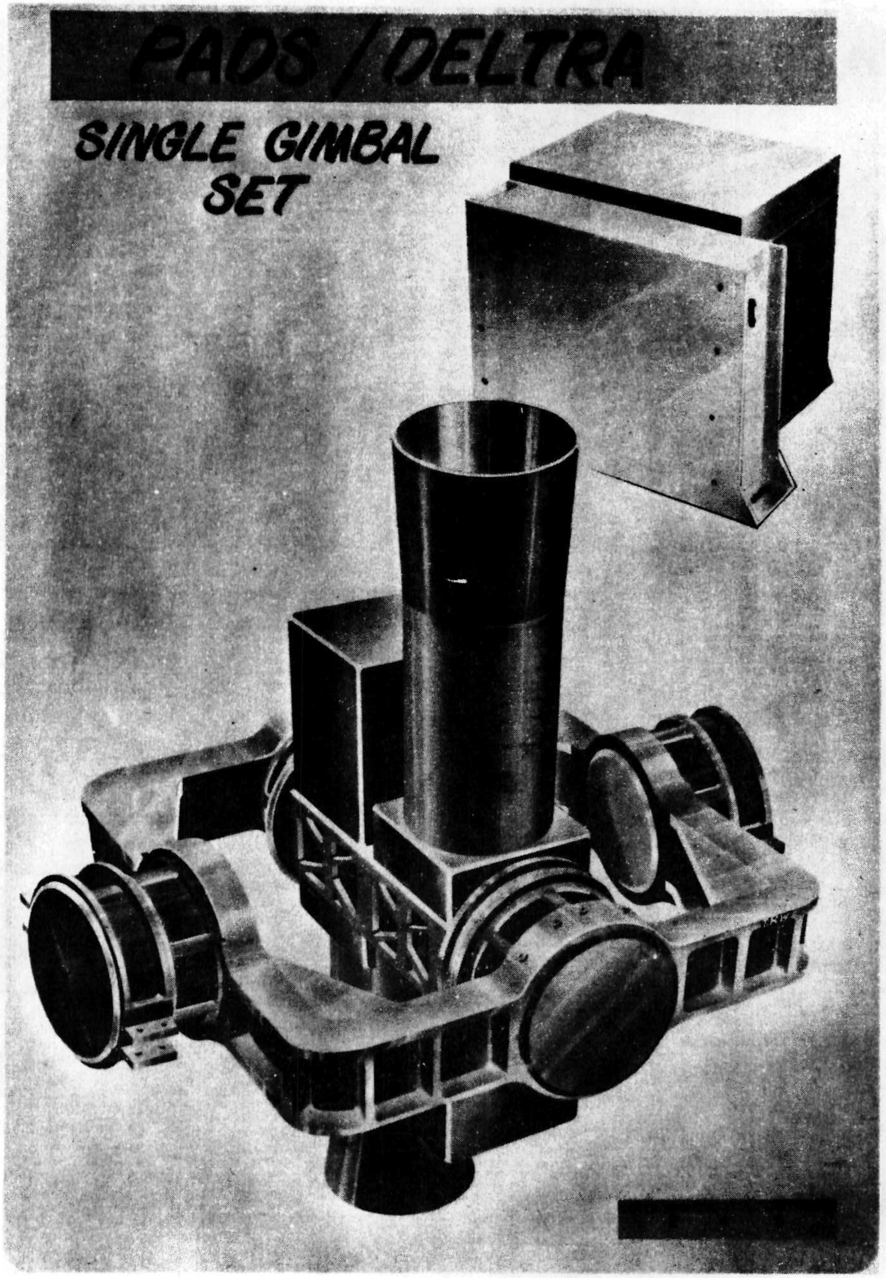


Figure 5-20. Combined Gimbal Candidate

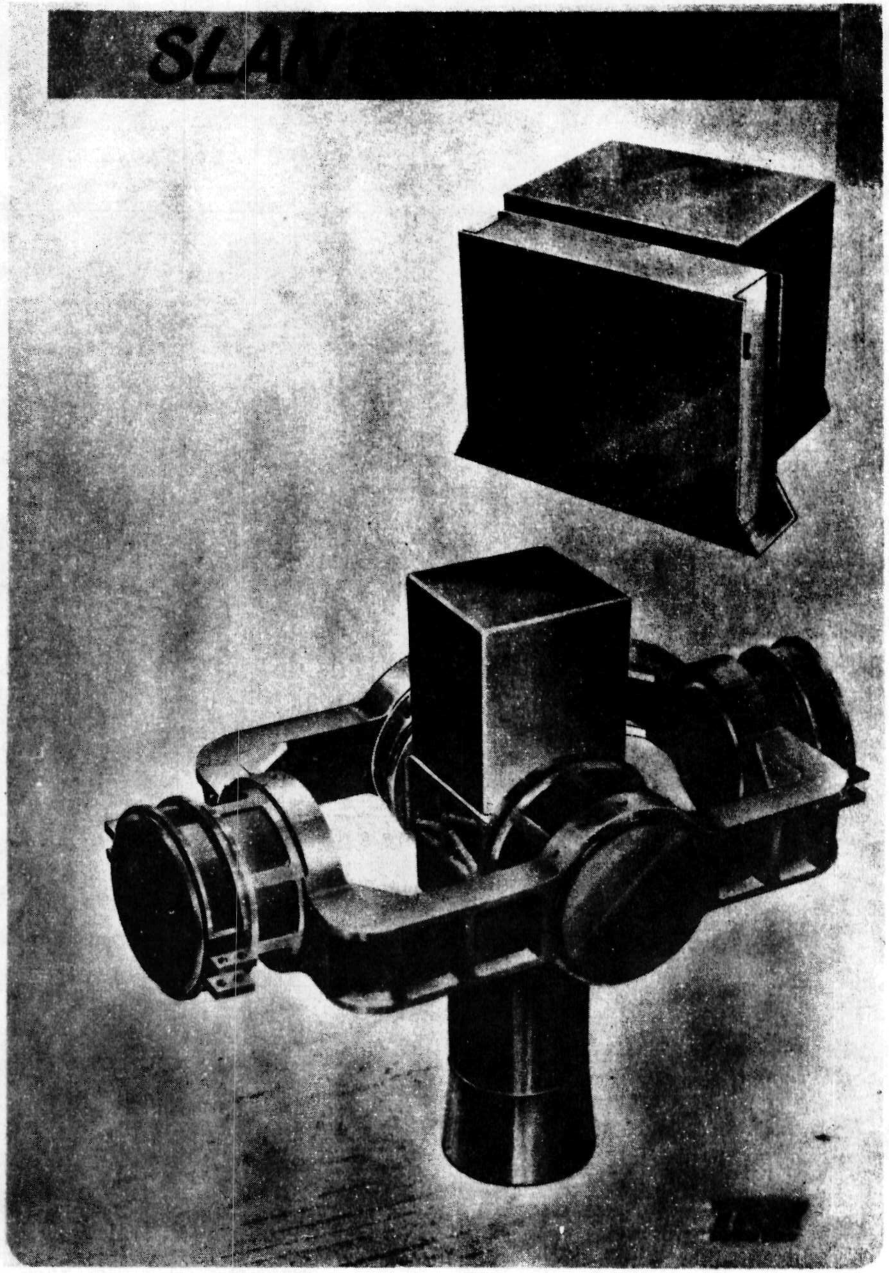


Figure 5-21. SLANT Candidate

Table 5-8. Landmark Tracker Mechanization Tradeoff

	Calibration	Estimated Recurring Cost ($000)	Spacecraft Interface (deg)	Feasibility (Status)	Weight (lb)	Power (w)	Reliability Estimates	Versatility
SPARS (SSAs + SITU)		$300	two 4 FOVs	Developed	40	22	0.96	Low altitude
DELTRA	5 axes	285	20 x 100	Designed	36	37	0.95	orbits
					76	59	0.91	
PADS (Tracker)		245	30 x 60	Partially developed	31	27	0.96	Low and medium orbits
DELTRA	6 axes	285	20 x 100	Designed	36	37	0.96	
					67	64	0.91	
Combined Gimbal	4 axes	340	30 x 60		57	44		Low and medium altitude orbits
Tracker			20 x 100	Designed			0.94	
SLANT	3 axes	300	30 x 60		43	37		Low and medium altitude orbits
			20 x 100	Designed			0.95	

Note: Reliability for 90 days, 50 percent duty cycle, 50 percent off failure rate.

The SLANT tracker provides an extremely versatile sensor through the combination of a wide sensitivity band and the large gimbal freedom. For example, the tracker can be used to track optical beacons (light sources) on the right side of an orbit through a combination of its star-tracker sensitivity and downward-look capability. Conceivably, it could be used for precise horizon-angle observations through the addition of appropriate scan and processing electronics; however, in this case, the wavelengths observed would be constrained by the detector's response. The use of two SLANT trackers should be considered for missions where high reliability is required. The duplication of a dual-function sensor results in dramatic improvements in system reliability.

COST ANALYSIS

A comparison of the potential cost of implementing each of the five candidate systems for the earth orbit navigation function required a model to account for the different areas in which expenditures will be necessary and to account for the various factors which can affect the cost. The specific elements considered in the cost model were:

- System DDT&E Cost
- Production Unit Cost
- Weight
- Power
- Ground Checkout & Maintenance of Flight Components
- Ground Support
- Reliability
- Software
- Ground Station Installation
- Ground Station Operation and Maintenance

The operational mission and program characteristics also affect the system cost through the number of scheduled missions and the duration of each mission. The two vehicles and programs considered were the space tug with 126 flights over a six year period and the space station with a mission length of ten years. The total functional requirements of each vehicle were considered in terms of the multifunctional capability of each candidate system. The functions considered (in addition to orbit navigation) were:

- Attitude reference
- Communications
- Rendezvous navigation
- Stationkeeping navigation

The implementation costs throughout the duration of each program were divided into two phases:

- Design, Development, Testing, and Engineering (DDT&E) - cost for system development.
- Investment and Operations (I&O) - cost associated with purchase of production systems and mission operations.

These two categories are distinct in that DDT&E costs are once only costs, and I&O costs are recurring costs depending heavily upon the number of missions and vehicles using the navigation system.

SUMMARY

Tables 6-1 and 6-2 summarize the relative program costs of implementing each of the five candidate systems for the space tug and space station programs respectively. The costs were computed relative to an arbitrary base system which was taken as the star/landmark system consisting of a combined star/landmark tracker (which also provides line-of-sight angle data for rendezvous and stationkeeping navigation) and a communications receiver/transmitter (which also provides ranging for rendezvous and stationkeeping navigation). Since a definite tradeoff exists for the rendezvous and stationkeeping navigation sensor (communication ranging addition combined with an optical tracker versus a separate rendezvous radar), the cost of implementing each candidate system with each rendezvous and stationkeeping navigation sensor was determined.

Inspection of Tables 6-1 and 6-2 show that the ground transponder system is the most expensive system to implement (relative to the other four candidate systems) for either rendezvous and stationkeeping navigation sensor on either the space tug or space station program. This is due primarily to the cost of ground station operation and maintenance and the poor reliability of both the onboard interrogator and the ground transponder resulting in a purchase of many production units and expensive ground checkout and system repair.

When the rendezvous radar is chosen as the rendezvous and stationkeeping navigation sensor, the least expensive system (of the five candidates) to implement on either the tug or space station would be the star/landmark system. This is due to the fact that the tracker for this system could obviously provide star and rendezvous target line-of-sight angle data and therefore only a communication ranging addition is necessary to provide all the required functions. As a result the other four systems are severely penalized by the cost of adding a separate rendezvous radar where the star/landmark tracker system is not. This separate rendezvous radar penalty is not only significant in the relative development cost but also very predominant on the tug because of the cost of weight.

For the case where a communication ranging addition combined with an optical tracker is chosen as the rendezvous and stationkeeping navigation sensor, the TDRS, MSFN and star/landmark systems are the least expensive and would cost approximately the same to implement on the space tug. The horizon sensor system would cost an additional 2-3 million dollars to implement on the tug. For the space station program the mission ground support costs for TDRS and MSFN become quite large due to the length of the program and the amount of navigation support required. As a result, the star/landmark tracker and the horizon sensor systems would be the cheapest to implement on the space station.

Table 6-1. Summary of Costs Relative to the Star/Landmark Tracker System for the Tug Program

SYSTEM / COST ELEMENT	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK TRACKER SYSTEM
DDT&E (13)	3.0 (-0.3)	2.5 (-0.8)	2.0 (-0.6)	2.5 (-0.8)	0 (0)
GROUND STATION INSTALLATION	NA	NA	0.1 (0.1)	NA	NA
SOFTWARE	0 (0)	-0.2 (-0.2)	0 (0)	-0.2 (-0.2)	0 (0)
SUBTOTAL	3.0 (-0.3)	2.3 (-1.0)	2.1 (-0.5)	2.3 (-1.0)	0 (0)
PRODUCTION UNITS (13)	0.7 (0)	0 (-0.6)	0.9 (1.4)	0 (-0.6)	0 (0)
WEIGHT (6)	5.7 (1.9)	3.7 (-0.1)	3.1 (1.8)	3.7 (-0.1)	0 (0)
POWER (8)	0.1 (0.1)	0 (0)	0.6 (0.7)	0 (0)	0 (0)
GROUND CHECKOUT AND SYSTEM REPAIR (9) (10)	0.6 (0.1)	0.4 (-0.1)	1.3 (1.1)	0.4 (-0.1)	0 (0)
RELIABILITY (12)	0.7 (0.6)	0.7 (0.6)	4.9 (9.2)	0.7 (0.6)	0 (0)
SUBTOTAL	7.8 (2.7)	5.0 (-0.2)	10.8 (14.2)	5.0 (-0.2)	0 (0)
MISSION GROUND SUPPORT	NA	2.8 (2.8)	NA	2.8 (2.8)	NA
GROUND STATION OPERATION AND MAINTENANCE	NA	-	0.6 (0.6)	-	NA
TOTAL	10.8 (2.4)	10.1 (1.6)	13.5 (14.3)	10.1 (1.6)	0 (0)

NOTES:
1. Parenthetical numbers are for a ranging (Com Δ) and an optical tracker for rendezvous; open numbers are for a radar for rendezvous.
2. All numbers are in millions of dollars.
3. Numbers under Cost Element give corresponding table number in Cost Analysis Section.

Table 6-2. Summary of Costs Relative to the Star/Landmark Tracker System for the Space Station Program

COST ELEMENT \ SYSTEM	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR LANDMARK TRACKER SYSTEM
DDT&E (13)	3.0 (-0.3)	2.5 (-0.8)	2.0 (-0.6)	2.5 (-0.8)	0 (0)
GROUND STATION INSTALLATION	NA	NA	0.1 (0.1)	NA	NA
SOFTWARE	0 (0)	-0.1 (-0.1)	0 (0)	-0.1 (-0.1)	0 (0)
SUBTOTAL	3.0 (-0.3)	2.4 (-0.9)	2.1 (-0.5)	2.4 (-0.9)	0 (0)
PRODUCTION UNITS (13)	0.6 (0)	0.3 (-0.2)	2.8 (2.5)	0.3 (-0.2)	0 (0)
WEIGHT (7)	0.1 (0)	0.1 (0)	0.3 (0.1)	0.1 (0)	0 (0)
POWER (8)	0.1 (0.2)	-0.1 (0)	0.8 (0.9)	-0.1 (0)	0 (0)
GROUND CHECKOUT AND SYSTEM REPAIR (10)	0.3 (-0.1)	0.3 (-0.1)	2.3 (1.8)	0.3 (-0.1)	0 (0)
RELIABILITY (11)	0.1 (0)	0.1 (0)	0.2 (0.3)	0.1 (0)	0 (0)
SUBTOTAL	1.2 (0.1)	0.7 (-0.3)	6.4 (5.6)	0.7 (-0.3)	0 (0)
MISSION GROUND SUPPORT	NA	6.3 (6.3)	NA	6.3 (6.3)	NA
GROUND STATION OPERATION AND MAINTENANCE	NA	-	1.0 (1.0)	-	NA
TOTAL	4.2 (-0.2)	9.4 (5.1)	9.5 (6.1)	9.4 (5.1)	0 (0)

NOTES:
1. Parenthetical numbers are for a ranging (Com Δ) and an optical tracker for rendezvous; open numbers are for a radar for rendezvous.
2. All numbers are in millions of dollars.
3. Numbers under Cost Element give corresponding table number in Cost Analysis section.

MULTIFUNCTION UTILIZATION CONSIDERATIONS

In order to perform a quantitative cost comparison of each system, the total functional requirements of the vehicle must be considered in terms of the multifunctional capability of each candidate system. As previously stated, the functions considered (in addition to orbit navigation) were:

- Attitude reference
- Communications
- Rendezvous navigation
- Stationkeeping navigation

The attitude reference function is assumed to necessitate an inertial reference assembly (either gimballed or strapdown) and a star tracker or mapper. The communication requirements are assumed to be spacecraft to ground, ground to spacecraft and spacecraft to spacecraft (shuttle to tug or space station and tug or space station to shuttle). For the rendezvous and stationkeeping navigation functions, target line-of-sight (LOS) angle data are assumed to be necessary as well as range or range and range rate. Each candidate navigation system can provide some support for these functional requirements if properly mechanized. For each candidate navigation system, the basic sensors required for the orbit navigation function and the minimal supplement of sensors required for the additional functions are listed in Table 6-3.

For the horizon sensor orbit navigation system, the basic sensor equipment consists of an attitude reference and horizon scanner(s). The optical tracker used for an attitude reference also could provide target LOS angle data, although tracking of the target may be a problem at long range (350-450 n.mi.) for rendezvous navigation. A receiver/transmitter must be added to provide communications and possibly long range ranging for rendezvous navigation and short range ranging for stationkeeping navigation. The tradeoff is 1) a separate rendezvous radar (which provides ranging and LOS angle data) versus 2) a communications receiver/transmitter which provides ranging combined with optical tracker LOS angle data for rendezvous and stationkeeping navigation.

The basic sensor equipment for the TDRS system consists of the communications receiver/transmitter including a range/range rate transponder and ground tracking and processing equipment. The TDRS range/range rate transponder does not provide range/range rate to the onboard computer and therefore could not be used to provide ranging for rendezvous and stationkeeping navigation. The communications receiver/transmitter could possibly provide long and short range ranging for rendezvous and stationkeeping navigation and the optical tracker which must be added to provide an attitude reference could possibly provide target LOS angle data as well. The tradeoff is the same as with the horizon sensor system, i.e., 1) a separate rendezvous radar versus 2) a communications receiver/transmitter which provides ranging combined with optical tracker LOS data.

Table 6-3. *Sensor Equipment Requirements for Multifunction Utilization**

SYSTEM / FUNCTION		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Orbit Navigation		Optical Tracker Horizon Sensor	$R/\dot{R}$ Xponder (Com Δ) GND Equipment	RCVR/Xmitter (A) GND Equipment	$R/\dot{R}$ Xponder (ComΔ) GND Equipment	Star/Landmark Tracker
Attitude Reference		-	Optical Tracker	Optical Tracker	Optical Tracker	-
Communication		RCVR/XMITTER	RCVR/XMITTER	RCVR/XMITTER (B)	RCVR/XMITTER	RCVR/XMITTER
Rendezvous Navigation	LOS Angles	-	-	-	-	-
	R or. R & R	Long R Ranging (Com Δ or RR)	Long R Ranging (Com Δ or RR)	-	Long R Ranging (Com Δ or RR)	Long R Ranging (Com Δ)
Station Keeping Navigation	LOS Angles	-	-	-	-	-
	R or. R & R	Short R Ranging (Com Δ or RR)	Short R Ranging (Com Δ or RR)	-	Short R Ranging (Com Δ or RR)	Short R Ranging (Com Δ)

*Inertial reference assembly (either gimballed or strapdown) assumed on all 5 systems.

The ground transponder orbit navigation system consists of an onboard receiver/transmitter and ground transponders. The basic orbit navigation receiver/transmitter would also provide the ranging and possibly the LOS angle data for rendezvous and stationkeeping navigation. In order to provide the additional functions considered, an optical tracker must be added for an attitude reference and possibly to measure LOS angles for rendezvous and stationkeeping navigation. A separate receiver/transmitter would be required for communication. For this system, the tradeoff is 1) a gimballed antenna for the basic orbit navigation receiver/transmitter versus 2) a gimballed optical tracker in order to obtain target LOS angle data for rendezvous and stationkeeping navigation.

The basic sensor equipment for the MSFN orbit navigation system consists of the communications receiver/transmitter including a range/range rate transponder and ground tracking and processing equipment. The same as with the TDRS system, the MSFN range/range rate transponder does not provide range/range rate to the onboard computer and therefore could not be used to provide ranging for rendezvous and stationkeeping navigation. Again, the communications receiver/transmitter could possibly provide long and short range ranging for rendezvous and stationkeeping navigation and the optical tracker which must be added to provide an attitude reference could possibly provide target LOS angle data as well. The tradeoff is the same as with the horizon sensor and TDRS systems.

For the star/landmark system, the basic sensor equipment consists of a star/landmark tracker (which also provides target LOS angle data for rendezvous and stationkeeping navigation). A receiver/transmitter must be added to provide communications as well as long range ranging for rendezvous navigation and short range ranging for stationkeeping navigation. There appears to be no tradeoff with this system.

Typical characteristics for the sensors required onboard for each candidate system and for the other functional requirements considered earlier are given in Table 6-4. The parenthetical numbers are those chosen for the sensor for the cost analysis.

MISSION AND VEHICLE CHARACTERISTICS

The operational mission and program characteristics also affect the system cost through the number of scheduled missions and the durations of the missions. The two vehicles and programs to be considered are the space tug and the space station. Reference 54 provides a traffic model which is appropriate for the space tug. The following data were taken from this reference:

- For the years 1979-1990, a total of 677 shuttle flights are listed; 225 required third stages (99 Agena and 126 tug) to accomplish satellite placement.

- For the years 1979-1984, the non-reusable Agena with the capability to take 2800 pounds to an equatorial geosynchronous orbit and the reusable Agena with 13,250 pound capability are to be used for satellite placement when the shuttle does not have the capability.

Table 6-4. *Onboard Sensor Characteristics*

EQUIPMENT / PARAMETER	WEIGHT (POUNDS)	POWER (WATTS)	MTBF (HOURS)	DDT&E COST (M$)	PRODUCTION UNIT COST (K$)
HORIZON SENSOR ASSEMBLY	17-45 (45)	10-38 (38)	20K-430K (100K)	0.5-1.0 (1.0)	50-200 (100)
TDRS TRANSPONDER (Addition to COM System)	(10)	(12)	(100K)	0.2-1.0 (0.5)	15-30 (25)
MSFN TRANSPONDER (Addition to COM System)	(10)	(12)	(100K)	0.2-1.0 (0.5)	15-30 (25)
GROUND TRANSPONDER SYSTEM					
OMNI	25-50 (50)	125-150 (125)	2K-9K (4K)	1.7 (1.7)	150-200 (170)
GIMBALLED	68 (85)	125-150 (125)	2K-9K (4K)	3-7 (7.0)	170-500 (300)
TRANSPONDER (GROUND)	20 (20)	100-120 (100)	6.5K-24K (15K)		13-75 (60)
STAR TRACKER	15-25 (20)	9-15 (12)	100K-250K (100K)	0 (0)	20 (20)
GIMBALLED FOR RENDEZVOUS	30-33 (32)	27-30 (28)	10K-50K (40K)	(2.7)	200 (200)
GIMBALLED FOR LANDMARKS	43 (43)	37 (37)	10K-35K (30K)	4 (4.0)	300 (300)
COM RANGING ADDITION FOR RENDEZVOUS	7 (7)	25 (25)	(100K)	(1.0)	(50)
RENDEZVOUS RADAR (GIMBALLED ANTENNA)	77 (85)	160 (125)	2K (4K)	(7.0)	(300)
LASER RADAR	28-35	25-30			

NOTE: Numbers in parentheses indicate actual numbers used in the cost analysis.

- Beginning in 1985 (thru 1990) the only third stage to be considered is the reusable tug, which can carry approximately 3500 pounds round trip (approximately 8000 pounds one way) to a equatorial geosynchronous orbit.

- Desired orbits of the NASA payloads ranged from 100 n.mi. circular to 38,646 circular with smaller elliptical orbits interspersed (DOD payload orbital parameters are classified).

- For all tug flights, the assumption was that the tug would return to rendezvous with the shuttle in the shuttle orbit and would take no longer than 7 days to complete the total mission (satellite placement and rendezvous).

- On-orbit assembly of payload to booster is allowed if required (may affect length of mission).

- For some payload placements (most of the planetary missions and the Applications Technology satellite) the tug is not capable of a round trip.

- The tug lifetime is 20 missions.

In addition to the tug, a space station mission will be considered with a mission length of 10 years.

An operating duty cycle is necessary to evaluate the cost contribution of power and reliability. For this purpose the following are assumed:

- The tug vehicle mission time is 7 days with the orbit navigation equipment operating one-third of the time (56 hours).

- The tug vehicle rendezvous navigation equipment operating time is 9 hours (1 rendezvous per tug mission).

- The space station mission time is 10 years with the orbit navigation equipment operating one-third of the time (29,200 hours).

- The space station rendezvous navigation equipment operating time is taken as 9 hours per rendezvous times one rendezvous per month (total of 1080 hours).

COST SENSITIVITIES

The sensitivities of total program cost to the various elements in the cost model are summarized in Table 6-5 for both the tug and space station. A discussion of each sensitivity together with supporting data or references is contained in the applicable section where the sensitivity is used.

Table 6-5. Summary of Program Cost Sensitivities

	TUG	SPACE STATION
WEIGHT		
• Launch	250 $/lb/flt	250 $/lb
• In-Orbit	200 $/lb/flt	NA
POWER	1.7 lb/KWH	1.0 lb/KWH
RELIABILITY		
• Value of Crew Time (for System Replacement)	NA	1900 $/hr
• Value of Unscheduled Down Time (In Orbit on the Vehicle)	NA	3800 $/hr
• Value Placed on a TUG Failure	5 M$	NA
GROUND SUPPORT		
• Personnel Cost	40 K$/yr	40 K$/yr
• Computer Cost	250 $/hr	250 $/hr
• MSFN Network	90 M$/yr	90 M$/yr
• TDRS Network	13 M$/yr	13 M$/yr
GROUND CHECKOUT & MAINTENANCE		
• Personnel Cost (DDT&E Phase)	20 $/hr	NA
• Personnel Cost (Production Phase)	15 $/hr	NA
SOFTWARE		
• Personnel Cost	40 K$/yr	40 K$/yr

WEIGHT

The weight of the navigation system is factored into cost by considering it as (1) an integral part of the tug booster system and (2) part of the shuttle payload. For the tug vehicle, the navigation system affects costs both as part of the shuttle payload and as part of the tug booster system. For the space station, the navigation system affects cost only as part of a payload. The following equation relates weight to cost for the tug mission:

$$C_{WT} = K \Delta W \, N_{TL}$$

where,

C_{WT} = total incremental value of weight for the tug relative to some arbitrary base system (dollars)

K = sensitivity of tug vehicle cost relative to weight

ΔW = incremental weight change relative to some arbitrary base system (pounds)

N_{TL} = number of tug vehicles launched

The sensitivity of tug vehicle cost relative to weight (K) includes all those costs associated with launch into low earth orbit and in-orbit operations (low earth orbit to geosynchronous orbit) for all program phases (i.e., DDT&E, Investment and operations). A value of 250 dollars/pounds is assigned as the cost of tug vehicle insertion into low earth orbit based on a 65,000 pound shuttle payload capability and 750 shuttle flights. The cost per pound for in-orbit operation of the tug was taken as 200 dollars/pound which is the average of the tug Phase A contractors study results for total recurring costs on the tug program. Therefore, the total cost sensitivity to weight for the tug (K) is 450 dollars/pound.

The relative weight penalties for the five orbit navigation systems considering the total functional requirements of the vehicle are given in Table 6-3 for both the separate rendezvous radar (RR) and the ranging (Com Δ) options. The weights shown are relative to an arbitrary base system which is defined as follows:

RR Option	Ranging (Com Δ) Option
star tracker	optical tracker
Com RCVR/XMTR	Com RCVR/XMTR
RR	Ranging (Com Δ)

Table 6-6. Tug Relative Weight Penalty

RR OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Star Tracker Com RCVR/Xmitter RR	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment	Base + Horizon Sensor	Base + R/Ṙ Xponder + GND Equip.	Base + RCVR/Xmitter + GND Equip. -RR	Base + R/R Xponder + GND Equip.	Base + Optical Tracker Δ -RR + Ranging (Com Δ)
Onboard Δ Weight Penalty (lbs.)	45 (100)	10 (65)	0 (55)	10 (65)	-55 (0)
Onboard Δ Weight Penalty (M$)	(5.7)	(3.7)	(3.1)	(3.7)	(0)

RANGING (COM Δ) OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Optical Tracker Com RCVR/Xmitter Ranging (Com Δ)	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment	Base + Horizon Sensor	Base + R/Ṙ Xponder + GND Equip.	Base + RCVR/Xmitter + GND Equip. - Ranging (Com Δ)	Base + R/Ṙ Xponder + GND Equip.	Base + Optical Tracker Δ
Onboard Δ Weight Penalty (lbs.)	45 (34)	10 (-1)	43 (32)	10 (-1)	11 (0)
Onboard Δ Weight Penalty (M$)	(1.9)	(-0.1)	(1.8)	(-0.1)	(0)

NOTE: Numbers in parenthesis indicate weight penalty relative to the star/landmark system.

For the relative costing purposes, the RR considered was a continuous wave, coherent system (similar to the LM RR) and not the laser radar of Table 6-2. The laser radar could provide the additional function of a docking sensor; however, its long range (300-400 n.mi.) rendezvous capability is questionable. Use of the laser radar instead of the CW radar would not significantly affect the results shown in Table 6-2.

For the separate RR option Table 6-2 shows that the star/landmark system incurs by far the smallest onboard weight penalty (relative to the other four systems). However, it should be noted that a separate RR is not included in the star/landmark system because the tracker for this system could obviously provide target LOS angle data as well and therefore only a ranging (Com Δ) system need be added to provide all the required functions. The horizon sensor system incurs the largest onboard weight penalty because the basic orbit navigation sensor cannot support the functions for rendezvous and stationkeeping.

When the ranging (Com Δ) option is chosen as the rendezvous and stationkeeping navigation sensor, the optical tracker must provide target LOS angle data and the communications receiver/transmitter must also provide ranging. For this option, Table 6-3 shows that the TDRS, MSFN and the star/landmark systems incur the smallest (and approximately the same) onboard weight penalty since for TDRS and MSFN only a range/range rate transponder must be added to the base system and for the star/landmark system only a small tracker addition must be made to provide all the functions required. The horizon sensor system again incurs the largest onboard weight penalty.

For the space station mission, the value of weight must be based upon the total number of systems requied for the ten year mission. The total systems required in orbit is determined in the reliability section and is listed in Table 6-11. The total weight into orbit required for the candidate systems for the two rendezvous options are given in Table 6-7. These relative weights are like those for the tug in that the landmark tracker system is much lighter for the rendezvous radar option, and the landmark tracker, TDRS, and MSFN are much lighter for the com delta option. The total value assigned for the space station system weight is:

$$C_{WS} = C_{SL} * WGT$$

where

C_{SL} is the cost per pound for insertion of the space station and its spares (dollars/pound).

WGT is the total weight required for the system.

The cost per pound for insertion of the space station and its spares and the cost per pound for insertion of the tug vehicle is the same since the shuttle vehicle is used for both. A value of 250 dollars/pound was previously assigned based on a 65,000 pound shuttle payload capability and 750 shuttle flights.

Table 6-7. Required Space Station Weight

RR OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

SYSTEM	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Weight (lbs)	790(554)	685(449)	1335(1099)	685(449)	236(0)
Cost (M$)	(0.14)	(0.11)	(0.27)	(0.11)	(0)

RANGING (COM Δ) OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

SYSTEM	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Weight (lbs)	252(16)	147(-89)	846(610)	147(-89)	236(0)
Cost (M$)	(0)	(-0.02)	(0.15)	(-0.02)	(0)

NOTE: Numbers in parenthesis indicate weight penalty relative to the star/landmark system.

POWER

The relative power requirements for the candidate systems were computed from the previously defined sensor characteristics and duty cycles. These requirements are given in Table 6-8 for the two alternate considerations for the rendezvous sensor.

Inspection of Table 6-8 shows that the ground transponder system incurs by far the largest onboard relative power penalty for both the tug and space station missions. This is due to utilizing equipment requiring relatively high power for long periods of time for orbit navigation. The star/landmark, TDRS, and MSFN systems incur the smallest onboard relative power penalty for both the tug and the space station.

The relative power penalty cost in terms of dollars is assessed for the five systems by using the equation relating weight to cost and employing a factor to relate kilowatt hours to pounds. The following factors were used in the cost analysis:

Tug: 1.7 pounds/KWH (typical shuttle equivalence)

SS: 1.0 pounds/KWH

It is expected that the actual cost penalty for the space station is somewhat less than the above figure. However, the 1 pound/KWH was used since the overall cost is relatively insensitive to this parameter.

GROUND CHECKOUT AND MAINTENANCE

Ground checkout and maintenance contributes to the total program cost by requiring men and test equipment to establish the flight readiness of each system between tug missions and to perform repair on failed systems for both the tug and space station. The large number of engineering man-hours expended during the DDT&E phase for ground test equipment installation and checkout, subsystem tests and flight vehicle tests are included in the system DDT&E cost.

The amount of time required to perform subsystem tests between tug missions is primarily a function of system complexity and the detailed testing philosophy. Table 6-9 shows the equipment involved (relative to the base system) and the estimated number of man-hours required for each of the five systems for both the separate rendezvous radar and the ranging (Com Δ) options. The relative cost in terms of dollars to perform the checkout and maintenance for the tug was determined using the following relationship:

$$C_{MT} = K_{MP} T_{MP} N_{TL}$$

where

C_{MT} = Tug ground checkout and maintenance cost relative to some arbitrary base system (dollars)

K_{MP} = Sensitivity of tug maintenance cost during the production phase relative to time (dollars/man hour)

Table 6-8. Relative Power Penalty

RR OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Star Tracker Com RCVR/XMTR RR		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment		Base + Horizon Sensor(s)	Base + R/Ṙ XPDR + GND Equip.	Base + RCVR/XMTR + GND Equip. -RR	Base + R/Ṙ XPDR + GND Equip.	Base + OPTL TRKR Δ -RR +Ranging (Com Δ)
TUG Onboard Δ Power Penalty	KWH	2.1 (1.4)	0.7 (0)	7.0 (6.3)	0.7 (0)	0.7 (0)
	M$	(0.13)	(0)	(0.61)	(0)	(0)
SS Onboard Δ Power Penalty	KWH	1110 (460)	350 (-300)	3650 (3000)	350 (-300)	650 (0)
	M$	(0.11)	(-0.07)	(0.75)	(-0.07)	(0)

RANGING (COMΔ) OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE OPTL TRKR Com RCVR/XMTR Ranging (Com Δ)		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment		Base + Horizon Sensor(s)	Base + R/Ṙ XPDR + GND Equip.	Base + RCVR/XMTR + GND Equip. - Ranging (Com Δ)	Base + R/Ṙ XPDR + GND Equip.	Base + OPTL TRKR Δ
TUG Onboard Δ Power Penalty	KWH	2.1 (1.6)	0.7 (0.2)	7.9 (7.4)	0.7 (0.2)	0.50 (0)
	M$	(0.15)	(0.02)	(0.71)	(0.02)	(0)
SS Onboard Δ Power Penalty	KWH	1110 (847)	350 (87)	3760(3497)	350 (87)	263 (0)
	M$	(0.21)	(0.02)	(0.87)	(0.02)	(0)

NOTE: Numbers in parenthesis indicate power penalty relative to star/landmark system.

Table 6-9. Tug Relative Ground Checkout and Maintenance of Onboard Systems Penalty

A. RR Option For Rendezvous and Stationkeeping Navigation Sensor

BASE STAR TRACKER COM RCVR/XMTR, RR		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment		Base + Horizon Sensor(s)	Base + R/Ṙ XPDR + GND Equipment	Base + RCVR/XMTR + GND Equipment -RR	Base + R/Ṙ XPDR + GND Equipment	Base + OPTL TRKR Δ -RR +Ranging (Com Δ)
Δ Ground Checkout and Maintenance Penalty (Equipment)		Horizon Sensor(s) C/O	R/Ṙ XPDR C/O	RCVR/XMTR C/O RR C/O	R/Ṙ XPDR C/O	OPTL TRKR Δ c/o -RR c/o +Ranging (Com Δ) c/o
Δ GND c/o and Maint. Penalty	Manhours/Flight	80(80)	12(12)	0(0)	12(12)	0(0)
	M$	(.2)	(0)	(0)	(0)	(0)

B. Ranging (COM Δ) Option for Rendezvous and Stationkeeping Navigation Sensor

BASE OPTICAL TRACKER COM RCVR/XMTR RANGING (COM Δ)		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment		Base + Horizon Sensor(s)	Base + R/Ṙ XPDR + GND Equipment	Base + RCVR/XMTR + GND Equipment - Ranging (Com Δ)	Base + R/Ṙ XPDR + GND Equipment	Base + OPTL TRKR Δ
Δ Ground Checkout and Maintenance Penalty (Equipment)		Horizon Sensor(s) C/O	R/Ṙ XPDR C/O + GND Equip. C/O	RCVR/XMTR C/O + GND Equip. C/O - Ranging (Com Δ) C/O	R/Ṙ XPDR C/O + GND Equip. C/O	OPTL TRKR Δ C/O
Δ GND c/o and Maint. Penalty	Manhours/Flight	80(32)	12(-36)	56(8)	12(-36)	48(0)
	M$	(.1)	(-.1)	(0)	(-.1)	(0)

NOTE: Numbers in parenthesis indicate ground checkout & maintenance penalty relative to the star/landmark system.

T_{MP} = Incremental ground checkout and maintenance time during the production phase relative to some arbitrary base system (man hours/flight).

N_{TL} = Number of tug vehicle flights.

The relative system repair penalty in terms of equipment and dollars for each of the five systems for both the tug and space station is shown in Table 6-10. The total number of repairs required for each flight sensor is determined in the reliability section and is listed in Table 6-11. It is assumed that all failures would be repaired at a cost of one-half the original production unit cost for each sensor.

It should be noted that repair of failed ground station transponders for the ground transponder system is also included in Table 6-10. The number of failed ground station transponders (and therefore the number of repairs required), the total number of replacements and the required number of units purchased for seven ground sites were determined assuming a replacement at an MTBF age and a continually operating system. The results were as follows:

	NO. OF REPLACEMENTS	NO. OF FAILURES	NO. OF UNITS PURCHASED
Tug Program (6 years)	37	25	19
SS Program (10 years)	62	41	28

The same as with the flight sensor repairs, it is assumed that all failures would be repaired at a cost of one-half the original production unit cost.

GROUND SUPPORT

Ground support as defined herein implies active real time participation by men and/or equipment on the ground in order to complete a given function. Two of the five candidate orbit navigation systems (TDRS and MSFN) incur ground support costs since an integral part of the navigation system is located on the ground. The ground transponder system incurs no ground support costs since the ground stations are assumed to operate in a passive role (maintenance of these ground stations are included in the ground checkout and maintenance section). The horizon sensor and star/landmark systems are completely autonomous and therefore incur no ground support penalty.

It is assumed that both the TDRS and MSFN systems will be operational during the space shuttle program and will be available as a result of the communications requirement. Therefore, the ground support cost directly attributable to the navigation function involves two factors: 1) ground data processing and computations and 2) additional MSFN ground tracking stations required for navigation above that required for communications. Assuming no ground software development costs are incurred (i.e., use

Table 6-10. Relative System Repair Penalty for the Tug and Space Station

A. RR Option for Rendezvous and Stationkeeping Navigation Sensor

BASE STAR TRACKER COM RCVR/XMTR, RR		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment		Base + Horizon Sensor(s)	Base + R/R XPDR + GND Equipment	Base + RCVR/XMTR + GND Equipment-RR	Base + R/R XPDR + GND Equipment	Base + OPTL TRKRΔ -RR + Ranging (ComΔ)
Tug Repair Penalty	Unit	3 RR repairs	3 RR repairs	4 RCVR/XMTR Repairs + 25 GND Unit repairs	3 RR repairs	No repairs
	M$	.4 (.4)	.4 (.4)	1.3 (1.3)	.4 (.4)	0 (0)
SS Repair Penalty	Unit	3 RR repairs	3 RR repairs	8 RCVR/XMTR repairs +41 GND unit repairs	3 RR repairs	1 LDMK TRKR repair
	M$	.4 (.3)	.4 (.3)	2.4 (2.3)	.4 (.3)	.1 (0)

B. Ranging (Com Δ) Option for Rendezvous and Stationkeeping Navigation Sensor

BASE OPTICAL TRACKER COM RCVR/XMTR RANGING (COM Δ)		HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment		Base + Horizon Sensor(s)	Base + R/R XPDR + GND Equipment	Base + RCVR/XMTR + GND Equipment - Ranging (Com Δ)	Base + R/R XPDR + GND Equipment	Base + OPTL TRKR Δ
Tug Repair Penalty	Unit	No repairs	No repairs	4 RCVR/XMTR repairs +25 GND unit repairs	No repairs	No repairs
	M$	0 (0)	0 (0)	1.1 (1.1)	0 (0)	0 (0)
SS Repair Penalty	Unit	No repairs	No repairs	8 RCVR/XMTR repairs + 41 GND unit repairs	No repairs	1 LDMK TRKR repair
	M$	0 (-.1)	0 (-.1)	1.9 (1.8)	0 (-.1)	.1 (0)

Note: Parenthetical numbers indicate penalty relative to the star/landmark system.

existing software) and assuming a ground computational facility were available (i.e., only incur computer costs for actual time used), the estimated ground data processing and computations costs for either MSFN or TDRS are as follows:

Tug only:	0.47 M$/yr (6 yrs)	= 2.8 M$
SS only:	0.63 M$/yr (10 yrs)	= 6.3 M$
Tug & SS:	0.63 M$/yr (10 yrs) + 0.11 M$/yr (6 yrs)	= 7.0 M$

In arriving at the above estimates, six navigation updates/day were assumed requiring 30 minutes computer time each. Three men/shift were assumed (3 shifts/day) which were shared during the six year overlap between the tug and space station program. If additional MSFN ground tracking stations are required for navigation above that required for communications a significant increase in the above figures would result for MSFN since the average cost is approximately 5 M$ per station per year.

RELIABILITY

The ability of a system to successfully perform its function during a mission depends upon the reliability of the system. The cost of a failure of equipment crucial to mission success is at least the cost of the one mission plus possibly the cost of loss or recovery of the vehicle (safety). Component redundancy within a system can provide the reliability goal of the system to successfully perform the mission. Functional redundancy (and operational procedures) may improve the probability of a successful mission, but more important, will improve the safety of the mission. The value which could be placed on a manned space station is very high, and the provision for failsafe functional redundancy will be assumed for all functions crucial to mission safety. For the tug vehicle, reliability will be considered as synonomous with safety.

The value of space station down time (time period the navigation equipment is inoperative) will depend upon the functional requirement while system is under repair. Scheduled maintenance can probably be performed during a period when there is no functional requirement; thus, incurring only the cost penalty for replacement.

An additional affect of reliability on a long term orbital mission is the scheduled inflight maintenance and the cost of providing spares. The short length of the tug mission precludes the requirement for scheduled maintenance and spares. For the space station application, the system failure rate and the shuttle to station traffic model will be used to determine the number of spares to be transported to the space station, the maintenance schedule, and the expected down time due to in-service failures.

Accurate failure characteristics of equipment are very difficult to obtain; particularly the lifetime characteristics equipment. For long duration missions the lifetime is important in determining the scheduled replacement policy.

The equipment is assumed to be past the burn in stage and to be operating in the region of constant failure rate until the lifetime is reached. The scheduled maintenance will replace the system before the lifetime is reached. Otherwise, the system is replaced at in-service failures.

For an exponential failure distribution the expected number of system replacements (scheduled and in-service failures) is given by:

$$N_R = \frac{1}{(1-e^{-\Delta T/\mu})^2} \left\{ \frac{T}{\mu} - \left(\frac{T}{\mu} + \frac{\Delta T}{\mu} - 1\right) e^{-\Delta T/\mu} - e^{-2\Delta T/\mu} \right.$$

$$\left. - \left(\frac{T}{\mu} + 1 - K\frac{\Delta T}{\mu}\right) e^{-K\,\Delta T/\mu} + \left[\frac{T}{\mu} + 1 - (K-1)\frac{\Delta T}{\mu}\right] e^{-(K+1)\,\Delta T/\mu} \right\}$$

where

$(K-1)\ \Delta T \leq T < K\ \Delta T,\ K = 1,2,\ldots$

ΔT is the time from installation to scheduled replacement (less than lifetime)

T is time period over which N_R is to be evaluated

μ is the mean time to failure.

This function is illustrated in Figure 6-1, and it can be seen that the number of replacements varies greatly with the ratios of mission duration and replacement age to mean time between failure. When the replacement age is greater than the mission duration, the function is simply the mission duration divided by the mean time between failure which is the expected number of in-service failures. Thus, this linear function gives the expected number of in-service failures and the other functions give the sum of the scheduled replacements and the in-serivce failures. Their difference is the expected number of scheduled replacements.

To evaluate the number of systems required for the space station, the system lifetime (replacement age) will be assumed to be the effective MTBF considering the one-third-on two-thirds-off operating duty cycle for the orbit navigation systems and one rendezvous per month for the rendezvous navigation systems. These duty cycles result in the following effective MTBF's for the space station:

Orbit Navigation $\qquad \dfrac{\mu}{1 - 2/3\ (1-1/K)}$

Rendezvous Navigation $\qquad \dfrac{K\mu}{1 + 1/80\ (K - 1)}$

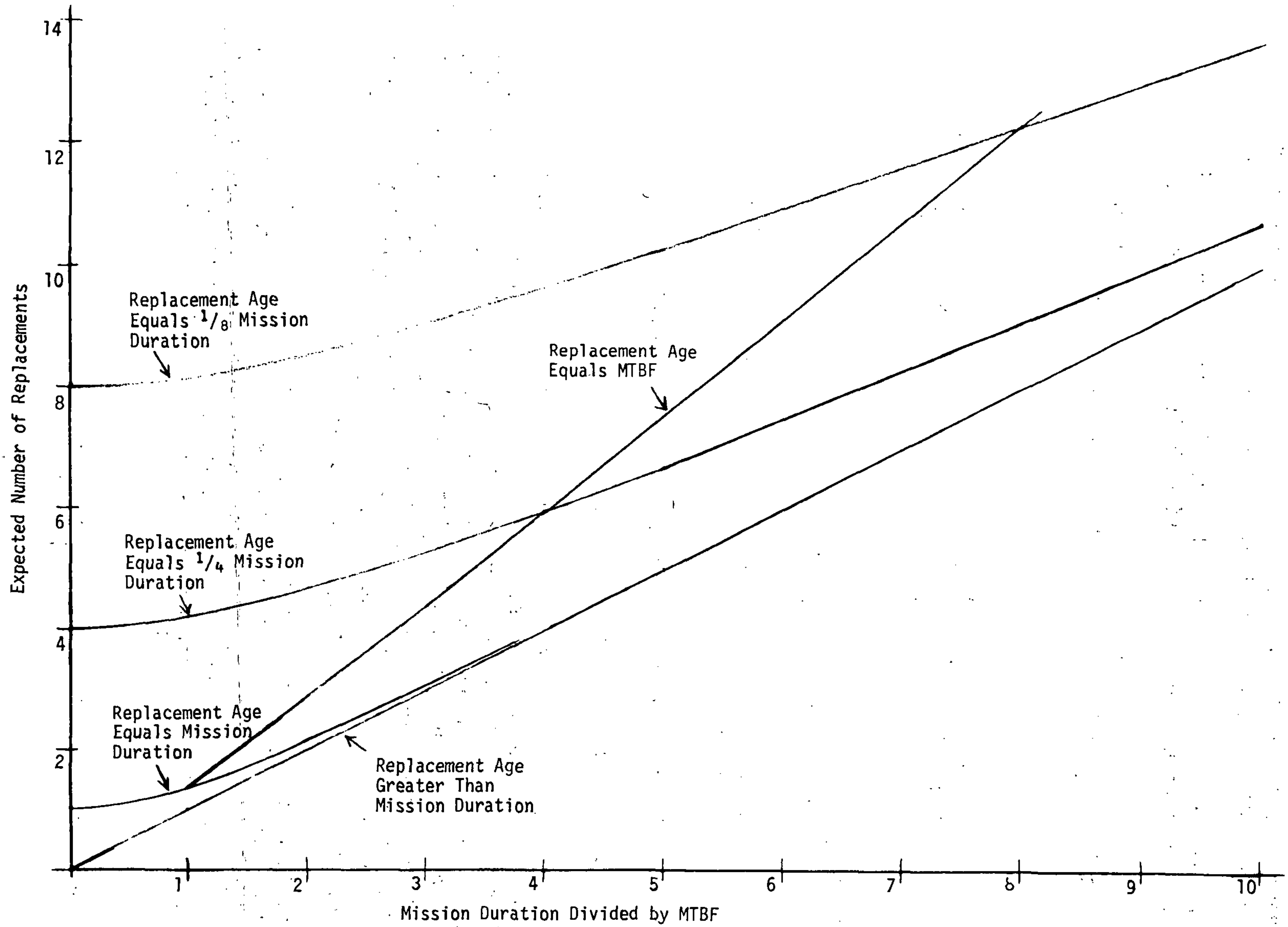


Figure 6-1. Expected Number of Replacements for Exponential Failure Distribution and Age or Failure Replacement Policy

The number of systems required for the space station is the sum of the original ship set, one remaining spare, and the expected number of replacements. The systems required for the space station was determined using a system off failure rate equal to one tenth of the system on failure rate (K=10) and are given in Table 6-11.

The value assigned to the space station for system replacements is determined from the following equation:

$$C = (N_R * CV + N_f * DTV) * RH$$

where

N_R is the number of replacements

CV is the value of the crew time required for replacement

RH is the crew hours required for replacement and checkout

N_f is the number of failures

DTV is the value of unscheduled down time

The value of the crew time for maintenance activities and the value of system down time are obtained from data of References 55 and 56. These data are:

Modular Space Station Operations Cost	\$361.2 m
Earth Orbital Experiments Operations Cost	\$591.9 m
RAM Operations Cost	\$ 61.3 m
Six Man Crew Man Hours Per Day	144
Man Hours Per Day for Vehicle Operations	25

The value for each man hour for operations is given by:

$$CV = \frac{\text{Total Operations Cost}}{\text{Total Operating Time}} \times \text{Portion of crew time for vehicle operations}$$

$$= \frac{9.844 \times 10^9}{8.75 \times 10^4} \left(\frac{25}{144}\right)$$

$$= 1900 \text{ \$/hr}$$

Table 6-11. Number of Systems Required

SENSOR	TUG SYSTEMS REQUIRED	EFFECTIVE MISSION MTBF/LIFETIME (YRS)	SPACE STATION REQUIREMENTS		
			REPLACEMENTS	FAILURES	TOTAL SYSTEMS
Star Tracker	9	28.6	0.3	0.3	3
Horizon Sensors	9	28.6	0.3	0.3	3
TDRS/MSFN/Com Ranging	9	28.6	0.3	0.3	3
Ground Transponder	9(repair 4)	1.2	13.0	8.3	7(repair 8)
Rendezvous Radar	9(repair 3)	3.6	4.1	2.8	4(repair 3)
Landmark Tracker	9	8.6	1.8	1.2	3(repair 1)
Optical Tracker	9	10.1	1.0	1.0	3

SYSTEM	SPACE STATION PENALTIES - RR OPTION/COM Δ OPTION		
	REPLACEMENTS	FAILURES	COST RELATIVE TO THE STAR/LANDMARK SYSTEM (M$)
Horizon Sensor	4.7/1.6 (2.6/-.5)	3.4/1.6 (1.9/.1)	.06/0
TDRS	4.7/1.6 (2.6/-.5)	3.4/1.6 (1.9/.1)	.06/0
Ground Transponder	13.3/14.0 (11.2/11.9)	8.6/9.3 (7.1/7.8)	.24/.26
MSFN	4.7/1.6 (2.6/-.5)	4.7/1.6 (1.9/.1)	.06/0
Star/Landmark	2.1/2.1 (0/0)	1.5/1.5 (0/0)	0/0

NOTE: Numbers in parenthesis indicate penalty relative to the Star/Landmark System

The cost for system (navigation or attitude reference) down time is the station operations cost per hour times the portion of time the system is required (same as operational duty cycle). This cost is:

$$DTV = \frac{9.844 \times 10^9}{8.75 \times 10^4} \left(\frac{1}{3}\right)$$

$$= 3800 \ \$/hr$$

The traffic model for the tug vehicle which was given in the introductory section indicated that the tug program would be over a six year period with a tug mission lifetime of 20 missions, and a total of 126 missions. From this data it is assumed that there are seven tug vehicles and each vehicle is expended in two years (average of one month between missions). The lifetimes of the candidate navigation systems should be such that no scheduled replacements are necessary. Using the operating duty cycle of 1/3 on - 2/3 off, 126 seven-day missions, seven tugs, and a tug life of two years, then the total orbit navigation system operating times are:

Mission On Time = 126 * 7/3 = 294 days

Mission Off/Standby = 126 * 14/3 = 588 days

Ground Off/Standby Time = 7 * 730 - 7 * 126 = 4228 days

For the rendezvous navigation systems, the operating times are:

Mission On Time = 47.2 days

Mission Off/Standby Time = 835.8 days

Ground Off/Standby Time = 4228 days

The final mission for each tug is assumed to be a one-way mission, thus requiring a new system for each tug vehicle. With the addition of one leftover spare, then the total number of systems required for the tug program including the failure replacements is given by:

$$N_T = 8 + \frac{\text{ON TIME}}{\mu} + \frac{\text{OFF TIME}}{K\mu}$$

where

μ is the MTBF for the system On

$K\mu$ is the MTBF for the system Off/Standby

The number of systems required for the tug program was determined using a system off failure rate equal to one tenth the system on failure rate (K=10) and is given in Table 6-11.

To assign a cost for the probability of a tug failure, the multifunctional usage of the candidate systems must be considered. The probability of a failure of a mission is given by:

$$P = 1 - P_S\ (ATT) * P_S\ (RN) * P_S\ (ON) * P_S\ (REM)$$

where

P_S is the probability of success for the functions: attitude reference, rendezvous navigation, orbit navigation, and all remaining functions.

The equipment to be considered for the candidate systems for the two alternate rendezvous navigation sensors are listed in Table 6-12. This table also contains the probability of a tug failure for the candidate systems using the effective MTBF of Table 6-11 for the orbit navigation systems, a P_S(REM) = 0.95, and an effective MTBF for the rendezvous navigation system of

$$\frac{K\mu}{1\text{-}3/56\ (K-1)}$$

The total cost assigned to tug failure is obtained from:

$$C = 126 * P * TMV$$

where

TMV is the value placed on a tug failure.

DDT&E AND PRODUCTION UNIT COST

The relative DDT&E and production unit cost for each of the five candidate systems are shown in Table 6-13 for each rendezvous navigation sensor option. The total production unit cost based on the required number of systems is also shown for both the tug and space station. The required number of systems was determined in the reliability section and is shown in Table 6-11 and the individual sensor development and production unit cost was tabulated in Table 6-4. For the ground transponder system the data in Table 6-13 includes the cost of the required number of ground units for seven ground sites which was determined in the ground checkout and maintenance section.

SOFTWARE

The cost assigned to software for the candidate systems is determined from the onboard computational requirements for each system. All systems

Table 6-12. Tug Failure Probability

RR OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Star Tracker, Com RCVR/XMTR, RR	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment for Reliability	Base + Horizon Sensor	Base + $R/\dot{R}$ XPDR + GND Equip.	Star tracker + RCVR/XMTR + GND Equip.	Base + R/R XPDR + GND Equip.	Landmark Tracker + Ranging (Com Δ)
Failure Probability	.0533(.0011)	.0533(.0011)	.0600(.0078)	.0533(.0011)	.0522(0)
Cost (M$)	(0.69)	(0.69)	(4.91)	(0.69)	(0)

RANGING (COM Δ) OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Optical, Tracker, Com RCVR/XMTR, Ranging (Com Δ)	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment for Reliability	Base + Horizon Sensor	Base + $R/\dot{R}$ XPDR GND Equip.	Base + RCVR/XMTR GND Equip. - Ranging (Com Δ)	Base + $R/\dot{R}$ XPDR GND Equip.	Base + Landmark Tracker - Optical Tracker
Failure Probability	.0531(.0009)	.0531(.0009)	.0668(.0146)	.0531(.0009)	.0522(0)
Cost (M$)	(0.57)	(0.57)	(9.2)	(0.57)	(0)

NOTE: Numbers in parenthesis indicate failure probability relative to the star/landmark system.

Table 6-13. Relative System DDT&E and Production Unit Cost

RR OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Star Tracker Com RCVR/Smitter RR	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment	Base + Horizon Sensor	Base + R/R Xponder + GND Equip.	Base + RCVR/Xmitter + GND Equip. -RR	Base + R/R Xponder + GND Equip.	Base + Optical Tracker Δ -RR + Ranging (Com Δ)
Relative DDT&E Cost(M$)	1.0 (3.0)	0.5 (2.5)	0 (2.0)	0.5 (2.5)	-2.0 (0)
Relative Prod. Unit Cost (K$)	100. (75)	25. (0)	60. (35)	25. (0)	25 (0)
Tug Total Prod. Cost Based on No. of System Required (M$)	3.8 (0.7)	3.1 (0)	4.0 (0.9)	3.1 (0)	3.1 (0)
SS Total Prod. Cost Based on No. of Systems Required (M$)	1.6 (0.6)	1.3 (0.3)	3.8 (2.8)	1.3 (0.3)	1.0 (0)

RANGING (COM Δ) OPTION FOR RENDEZVOUS AND STATIONKEEPING NAVIGATION SENSOR

BASE Optical Tracker Com RCVR/Xmitter Ranging (Com Δ)	HORIZON SENSOR SYSTEM	TDRS	GROUND TRANSPONDER SYSTEM	MSFN	STAR/LANDMARK SYSTEM
Total Equipment	Base + Horizon Sensor	Base + R/R Xponder + GND Equip.	Base + RCVR/Xmitter + GND Equip. - Ranging (Com Δ)	Base + R/R Xponder + GND Equip.	Base + Optical Tracker Δ
Relative DDT&E Cost(M$)	1.0 (-0.3)	0.5 (-0.8)	0.7 (-0.6)	0.5 (-0.8)	1.3 (0)
Relative Prod. Unit Cost (K$)	100. (0)	25. (-75)	180. (80)	25. (-75)	100. (0)
Tug Total Prod. Cost Based on No. of Systems Required (M$)	3.1 (0)	2.5 (-0.6)	4.5 (1.4)	2.5 (-0.6)	3.1 (0)
SS Total Prod. Cost Based on No. of Systems Required (M$)	1.0 (0)	0.8 (-0.2)	3.5 (2.5)	0.8 (-0.2)	1.0 (0)

NOTE: Numbers in parenthesis indicate costs relative to the star/landmark system.

will have the software for functional requirements such as:

- Attitude Reference Updating
- Trajectory Propagation
- Rendezvous Navigation Updating
- Universal Pointing Vectors

Because the MSFN and TDRS require no additional onboard software, the requirements of the ground transponder, horizon sensor, and unknown landmark tracker systems will be determined relative to MSFN and TDRS.

Memory Requirements

Estimates of program size are difficult to make and are of marginal reliability until detailed assumptions are made concerning the hardware characteristics and the accuracy requirements of the navigation system. The computer parameters affecting program size are word length, cycle time, and the computer's instruction set.

For the problem under consideration, the memory requirements for the three schemes providing onboard orbital navigation capability will probably exceed the requirements of the other two schemes by not more than 20%. This assumes that most of the software required for the rendezvous navigation function may be modified for orbital navigation use with little penalty.

Capital and operating costs for the computer hardware will thus not vary significantly among the five navigation schemes. However, capital and operating costs for the software development, maintenance, and operational support may have significant differences.

Software Development and Support

The costs incurred in the development, maintenance, and operational support of the onboard software will be expressed in terms of manyears. The estimates given assume that the work is done in orderly fashion by experienced personnel with access to an existing inventory of related computer programs. Costs could easily run two to five times the estimates given in other circumstances.

"Development" includes formulation of the onboard computer programs, verification that the adopted formulation is adequate via simulations, coding of the programs in the onboard language, and verification of the program integrity. Development costs are one time expenditures.

"Operation" includes provision of mission-specific data (required by simplifications in the onboard code), and making pre-mission simulations to verify that navigation coverage is adequate to achieve navigation accuracy requirements.

It is assumed that the work is done within the context of a previously defined software executive system for the onboard computer, and that methods for simulating the onboard computer and compiling programs for the computer have already been developed.

Common Software

- Attitude Reference Control and Updating

  Development: 2.5 manyears

  Operation: 0.1 manyears/year

- Trajectory Propagation

  Development: 2.0 manyears

  Operation: 0.0 manyears/year

- Rendezvous Navigation Filter

  Development: 3.0 manyears

  Operation: 0.5 manyears/year

- Universal Pointing Vectors

  Development: 1.5 manyears

  Operation: 0.0 manyears/year

Horizon Sensor Navigation Software

Development costs for the horizon sensor orbital navigation software are relatively higher than for the other options, but operation costs are lower.

Extra development effort is required to formulate a usable onboard model of the horizon errors and to account for these errors in the navigation filter. The performance of the adopted formulation under a variety of conditions would be examined.

Operating costs will be low, however, because the availability of horizon measurements does not depend upon the ground track. Thus, little or no mission-specific simulation will be required.

Development: 4.0 manyears

Operation: 0.1 manyears/year

Ground Transponder Navigation Software

Development costs for the ground transponder orbital navigation software are moderate. Some extra effort is required to verify that the assumed pattern of ground transponders will provide adequate coverage for representative missions.

Operating costs will be larger than for the horizon sensor, however, because mission-specific simulations will be desired to establish the amount of tracking available for each mission:

Development: 3.0 manyears

Operation: 0.5 manyears/year

Unknown Landmark Tracker Navigation Software

Development costs for the unknown landmark tracker navigation software are moderate. The navigation filter is more complex than for the ground transponder software, but less effort is required for verification that coverage will be adequate.

Operating costs will be moderate. Some mission-specific planning and simulation will be required, but less than for the ground transponder method.

Development: 3.0 manyears

Operation: 0.3 manyears/year

Total Software Cost

The software cost relative to MSFN and TDRS for the six year tug program and the space station program are given below based upon a cost of $40,000 per manyear.

		HORIZON SENSOR SYSTEM	GROUND TRANSPONDER SYSTEM	UNKNOWN LANDMARK
Tug:	Development	160,000	120,000	120,000
	Operational	24,000	120,000	72,000
	TOTAL	200,000	240,000	192,000
SPACE STATION (Development)		160,000	120,000	120,000

REFERENCES

1. "Shuttle Tracking Systems Data Packages," NASA MSC/TCSD Document No. 1902, November 1971 (with errata).

2. Eggert, D., J. Godwin, and F. Joyce, "Parametric Analysis of RF Communications and Tracking Systems for Manned Space Stations," Hughes Aircraft Company Report No. SSD 00596R, January 1971.

3. Lynch, A. J., and F. A. Hartshorne, "Communications for a Manned Space Station," AIAA Paper No. 69-1079, AIAA 6th Annual Meeting and Technical Display, Anaheim, California, 20-24 October, 1969.

4. Ferry, W. W., "Orbit Navigation Performance for Candidate Space Station Systems," TRW IOC No. 7152.EON-010, 12 June 1972.

5. "Phase A Study Final Report, Volume 1, GSFC Mark 1 Tracking and Data Relay Satellite (TDRS) System Concept," Goddard Spaceflight Center, November 1969.

6. Sirri, N., C. Gilchrist, M. G. Dietl, et. al., "Final Study Report, Tracking and Data Relay Satellite Network (TDRSN)," Jet Propulsion Laboratory Report No. 760-40, 30 September 1969.

7. Earl, J. E., "Performance Requirements and Guidelines for Earth Orbit Navigation Study," TRW IOC No. 72-7152-EON-008, 28 April 1972.

8. Robinson, L. H., Jr., "PN Sequence Synchronization Analysis," TRW IOC No. 71-7151-LHR-73, 14 May 1971.

9. Final Technical Report, "CR-100 Implementation Study for Space Shuttle," Cubic Corporation, prepared for Intermetrics, Inc., 24 June 1971.

10. R. J. Panneton, "Earth-Orbital Navigation Systems," TRW IOC No. 71-7251-RJP-319, 18 February 1971.

11. R. J. Panneton, "General Operation of CIRIS Radio Reference Subsystem," TRW IOC No. 71:7151.RJP-382, 2 September 1971.

12. Motorola Inc., "AROD Application," prepared for NASA-MSFC under Contract No. NAS 8-11835.

13. O. Imai, "Possible Utilization of Ground Transponders for SSV Navigation," TRW IOC No. 7224.12-151, 4 September 1970.

14. L. J. Miller, "Orbital Range and Range Rate Errors Due to Tropospheric Refraction," TRW IOC No. 72:7152.5-19, 23 May 1972.

15. J. W. Muehlner, "Survey of RF Ranging Systems for Space Shuttle NAVAID Application," MSC/TCSD Document 1B2047, 10 May 1971.

REFERENCES (continued)

16. Cubic Corporation, "Space Shuttle Landing - Guidance System", Document No. P-70036A prepared for McDonnell Douglas Astronautics Company, 29 June 1970.

17. J. P. Gilmore, "A Non-Orthogonal Gyro Configuration," MIT/IL Report T-472, January 1967.

18. I. J. Williams, et al, "Autonomous Navigation Technology Contract Definition Phase Final Report (U)," TRW INC, May 1972, prepared for Department of the Air Force (SECRET).

19. Muller, E., and Philliou, P., "Status Report on Shuttle Ground Beacon Orbit Navigation Analysis," MIT Memo 23A STS No. 61-71, 8 December 1971.

20. Muller, E., "Orbit Navigation Performance," MIT Briefing Report 23486, January 1972.

21. "Horizon Definition Study Summary - Part 1," NASA CR-66180, Honeywell, Inc., October 1966.

22. Manry, C., "Preliminary Study Results of the Earth's Infrared Horizon Profile," MSC IN EG-70-48, 18 November 1970.

23. "Autonomous Navigation Techniques for Space Station," Final Study Report, North American Rockwell Corporation, Space Division, Report No. 192-100-CWR-71-437, 16 November 1971.

24. "Tracking and Data Relay Satellites by Stampl and Jones," IEEE Transactions on Aerospace and Electronic Systems, Vol. AES-6, No. 3, May 1970.

25. Douglas Keach, The Mathematical Physics Branch and James Knoedler, TRW, "A Study of the Feasibility of Selected Systems for Shuttle Navigation Support, Volume II - The Tracking Data Relay Satellite System," MSC IN 71-FM-51, 5 May 1971.

26. E. Muller, P. Kachmar, P. Philliou, "Report on Orbit Navigation Studies for the SSV," MIT STS Memo 24-70, 6 June 1970.

27. "Ground Network Considerations for Shuttle," PM-MO-MGR, George C. Marshall Space Flight Center, 12 November 1970.

28. Bradford, L. H., "Advanced Tracking Systems Study and Analysis Final Report," TRW Letter 5522.3-71-01, 28 January 1971.

29. Hendrickson, Howard T., "A Study of an Autonomous Unknown Landmark Tracking Navigation System," Aerospace Corp. Report - TR-059(6311)-4, February 1971.

REFERENCES (Continued)

30. L. Huggins, "Multipath Interference to Range and Doppler Measurement Errors," TRW Report 17618-H233-R0-00, November 1971.

31. "Earth Orbit Navigation System Study," TRW Proposal No. 21474.000, 29 November 1971.

32. "Modular Space Station, Phase B Extension, Preliminary Performance Specification, Volume I - Initial Station Systems," North American Rockwell Document SD 71-215-1, 3 December 1971.

33. "Modular Space Station, Phase B Extension, Preliminary System Design, Volume IV - Subsystems Analyses," North American Rockwell Document SD 71-217-4, January 1972.

34. "European Space Tug - Pre-Phase Study, Part 2, Study Digest," Hawker-Siddeley Dynamics Limited Document HSD TP 7264A, 1971.

35. MSC Internal Note MSC-EG-70-36, "Space Station Guidance, Navigation, and Control Systems Requirements, Revision 1," July 15, 1971.

36. NASA Memo FM 81 (72-19), "Shuttle Rendezvous Navigation Range and Range Rate R-quirements," February 3, 1972.

37. MBB Group Document, "Space Tug System Study, Pre-Phase A/Ext.," undated.

38. North American Rockwell Internal Letter No. 192-100-CWR-71-437, "Final Study Report - Autonomous Navigation Techniques for Space Station," November 16, 1971.

39. McDonnell Douglas Document MDC G2581, "Space Station Subsystems Preliminary Design Summary," November 1971.

40. McDonnell Douglas Document MDC G0634, "Space Station Preliminary Systems Design Data, Volume 1, Book 4, Utility Services," July 1970.

41. North American Rockwell Document SD 70-155-3-3, "Solar-Powered Space Station Definition, Volume III - Subsystem Definition, Part 3, Guidance and Control and Reaction Control Subsystems," July 1970.

42. North American Rockwell Document SD 70-159-3, "Solar-Powered Space Station, Preliminary Design, Volume III, Information Subsystem and Guidance and Control Subsystem," July 1970.

43. McDonnell Douglas Document MDC G0605, "Space Station Definition, Volume V, Subsystems, Book 5, Utility Services," July 1970.

44. AIAA Paper No. 68-449, "Synchronous Satellites as Part of a Complex Tracking Network," AIAA Conference, San Francisco, April 8-10, 1968.

45. North American Rockwell Document SD 70-547, "Guidance and Navigation Techniques for Long Duration Manned Space Systems," November 1970.

46. McDonnell Douglas Document MDC G0633, "Space Station Preliminary Performance Specifications, Volume 1," July 1970.

47. A. Frew, D. Kirby, P. Wheeler, and T. Huber, "An Integrated System for Precision Attitude Determination and Control," AIAA Paper No. 71-962 presented at the 1971 Guidance and Control Conference.

48. "Proposal for Strapdown Star Tracker-Space Shuttle Program," TRW No. 20622, May 1971.

49. "Quantic Industries Mod IV High-Accuracy, High Reliability Horizon Sensor System," Quantic Industries Report ETD-321A, April 1970.

50. "Proposal for Earth Horizon Sensor-Space Shuttle Program," TRW No. 20560, May 1971.

51. "A Functional Description of a Proposed Optical Subsystem for the Space Shuttle Vehicle," NASA/MSC Internal Note MSC-EG-71-1, May 1971.

52. "Multi-Mode Optical Sensor Design Study," NASA/MSC Internal Note MSC-EG-71-16, April 1971.

53. "Precision Earth Pointing System (PEPSY) Phase 1B Final Report," SAMSO TR 70-433, 12 December 1970 (SECRET).

54. "NASA/DOD Earth Orbit Shuttle Traffic Model in Support of the March 1972 Request for Proposal," MSC-06746, 21 March 1972.

55. "Modular Space Station, Phase B Extension, Extension Period Executive Summary," North American Rockwell Document SD 71-214, January 1972.

56. "Modular Space Station, Phase B Extension, Preliminary System Design, Volume II: Operations and Crew Analyses," North American Rockwell Document SD 71-217-2, January 1972.